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MSD 1536

RETURN TO
Historical Research Division
ASI/HOA
Maxwell AFB, AL 36112

K243.8636-36
V. 2, PT. 1
1 MAR 1959

2 7 OCT 1989

EXEMPTED FROM 25 MAR 1996
DECLASSIFICATION IAW EO 12958
REVIEW DATE______ REVIEWER (6)
REFER TO ______
~~EXEMPTION (S)~~: 1 2 3 ④ 5 6 7 8 9

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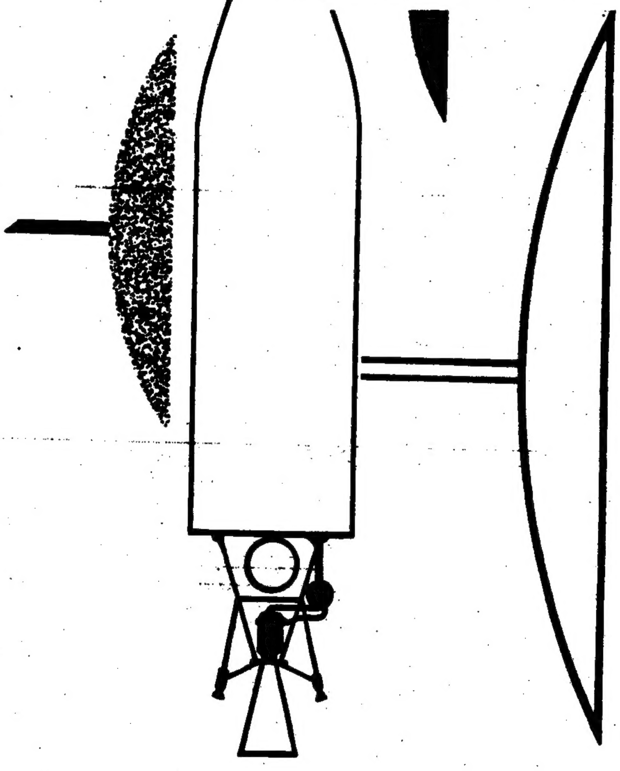

Pied Piper
DEVELOPMENT PLAN

VOL. II SUB-SYSTEM PLAN

A. Airframe

Excluded from General Declassification Schedule

DOWNGRADED AT 12 YEAR
~~INTERVALS; NOT AUTOMATICALLY~~
DECLASSIFIED. DOD DIR 5200.10

3-6806-2
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LOCKHEED AIRCRAFT CORPORATION
MISSILE SYSTEMS DIVISION
VAN NUYS, CALIFORNIA

MICROFILMED BY [illegible]

56RDZ_6524

THIS DOCUMENT CONTAINS INFORMATION AFFECTING THE NATIONAL DEFENSE OF THE UNITED STATES WITHIN THE MEANING OF THE ESPIONAGE LAWS, TITLE 18 U.S.C., SECTIONS 793 & 794. ITS TRANSMISSION OR THE REVELATION OF ITS CONTENTS IN ANY MANNER TO AN UNAUTHORIZED PERSON IS PROHIBITED BY LAW.

FOREWORD

The Advanced Reconnaissance System (ARS) consists of a satellite vehicle containing equipment to perform visual, ferret, and infrared reconnaissance, together with the necessary system of ground stations and data processing centers.

This Development Plan for the accomplishment of the ARS was prepared by the Missile Systems Division, Lockheed Aircraft Corporation and its subcontractors, CBS Laboratories and Eastman Kodak Company. The specifications for the system were determined in the course of a one-year study now being conducted for the United States Air Force under contract AF 33(616)-3105. The plan is presented in two parts; Volume I, System Plan, and Volume II, Subsystem Plan. The subsystems are described in separate books, Volume II-A through II-L.

PIED PIPER DEVELOPMENT PLAN

VOLUME I. SYSTEM PLAN

VOLUME II. SUBSYSTEM PLAN

A. Airframe
B. Propulsion
C. Auxiliary Power
D. Guidance and Control
E. Visual Reconnaissance
F. Electronic Reconnaissance
G. Infrared Reconnaissance
H. Vehicle Electronics
I. Airborne Test Systems
J. Vehicle Intercept and Control Ground Station
K. Ground Data Processing
L. Vehicle Ground Support

CONTENTS

Subsystem A Airframe

RDB PROJECT CARD (Form DD 613)

- Tab 1 General Design Specifications
- Tab 2 Subsystem Summaries
  - Milestones
  - Hardware Delivery
  - Test Schedules
  - R and D Schedules
- Tab 3 R and D Tests (Form ARDC 105)
- Tab 4 R and D Test Aircraft (Form ARDC 106)
- Tab 5 R and D Materiel (Form ARDC 107)
- Tab 6 Required Facilities
- Tab 7 R and D Contract Funds
- Tab 8 Estimate of Manpower Requirements

APPENDIX

1. Airframe
2. Vehicle Performance
3. Aerodynamics
4. Structures
5. Structural Analysis
6. Weight and Balance
7. Temperature Control of Satellite Components

References

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SECURITY CLASSIFICATION

MSD 1536

RDB PROJECT CARD

TYPE OF REPORT: New System - Delelopment Plan

REPORTS CONTROL SYMBOL DD-RDB(A)48

1. PROJECT TITLE
AIRFRAME SUBSYSTEM
FOR ADVANCED RECONNAISSANCE SYSTEM
(UNCLASSIFIED)

(PIED PIPER)

2. SECURITY: ~~S~~

3. PROJECT NUMBER: 1115

4. INDEX NUMBER:

5. REPORT DATE: 1 March 1956

6. BASIC FIELD OR SUBJECT:

7. SUBFIELD OR SUBJECT SUBGROUP:

7A. TECH. OBJ.:

8. COGNIZANT AGENCY:

9. DIRECTING AGENCY:

OFFICE SYMBOL:

TELEPHONE NO.:

10. REQUESTING AGENCY:

11. PARTICIPATION, COORDINATION, INTEREST:

12. CONTRACTOR AND/OR LABORATORY:
Missiles System Division
Lockheed Aircraft Corp.

CONTRACT/W.O. NO.: AF33(616)-3105

13. RELATED PROJECTS:

14. DATE APPROVED:

15. PRIORITY:

16.

17. EST. COMPL. DATES

RES.

DEV.

TEST

OP. EVAL.

18. FY	FISCAL ESTS. (M $)

19.

20. REQUIREMENT AND/OR JUSTIFICATION

20. This subsystem, the airframe of the satellite vehicle which is to be an additional stage to the WS-107 vehicle, must be capable of establishing the reconnaissance equipment in orbit and be compatible with the operation of the orbiting components of the reconnaissance systems. The operational environment is sufficiently different from that of any other system that no equivalent vehicle is available or under development.

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22. RDB	SN	CN	P	X.	I.	C.

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MISSILE SYSTEMS DIVISION 2.

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R&D PROJECT CARD
CONTINUATION SHEET

MSD 1536

1. PROJECT TITLE SUBSYSTEM DEVELOPMENT PLAN - AIRFRAME FOR ADVANCED RECONNAISSANCE SYSTEM (UNCLASSIFIED) (PIED PIPER)	2. SECURITY OF PROJECT ~~Secret~~	3. PROJECT NUMBER 1115
	4.	5. REPORT DATE 1 March 1956

21 a. **Brief Characteristics**

The airframe is the backbone of a final stage for the WS-107 vehicle, it has structural integrity for the boost phase, and carries the equipment required to establish itself in orbit, as well as furnish a framework for the components of the reconnaissance subsystems.

21 b. **Approach**

Booster vehicles will be derived from the WS-107 (Atlas program). The Pioneer vehicle airframe is designed to be compatible with the Atlas as well as the ground handling and launching equipment of the Atlas system. The Advanced vehicle airframe will be designed for use with a modified booster and launching system. Both airframes must satisfy the structural requirements of the boost phase, and contain components which make them consonant with the orbital environment. They must be capable of being stabilized in the orbit and also furnish an adequate framework during boost.

The tasks of the airframe subsystem, which are each described in par. 21c below, are:

1. Pioneer airframe structure
2. Pioneer adapter and separations
3. Pioneer pressurization system
4. Pioneer environment control
5. Advanced airframe structure and mechanisms
6. Booster modification for the advanced system
7. Advanced pressure system
8. Advanced environment control
9. Bola and other special configurations

21 c. **Tasks of the Subsystem**

1. **Pioneer Airframe Structure**

The airframe of the Pioneer vehicle is designed to be compatible with an unmodified Atlas System (WS-107). It is to house the Pioneer propulsion system and reconnaissance equipment. The airframe is approximately 61 inches in diameter and 18 feet in overall length. It is designed to tolerate about 8g longitudinal acceleration and 1.2g lateral acceleration. Vibration modes will be made compatible with those of the Atlas. The airframe structure

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R&D PROJECT CARD
CONTINUATION SHEET

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1. PROJECT TITLE SUBSYSTEM DEVELOPMENT PLAN - AIRFRAME FOR ADVANCED RECONNAISSANCE SYSTEM (UNCLASSIFIED) (PIED PIPER)	2. SECURITY OF PROJECT ~~Secret~~	3. PROJECT NUMBER 1115
	~~4.~~	5. REPORT DATE 1 March 1956

carries a destruct system which will not permit the vehicle to re-enter dense atmosphere. The configuration and structure of the Pioneer Vehicle is discussed in the appendix.

2. **Pioneer Adapter and Separation System Task**

No modifications of the Atlas are required for the Pioneer vehicle. Attachments to the present mounting ring will carry the adapter skirt which will include the separation system.

3. **Pioneer Pressurization System**

The Pioneer pressure system is a simple helium system providing pressure to the main propellant tanks and to any components that require a pressurized atmosphere. The principal reason for pressurizing the propellant tanks is to drive the propellant constituents into the combustion chamber at a pressure sufficiently high to make turbo pumps unnecessary. Furthermore, the internal tank pressures provide gas to any payload component that may require a pressurized environment. The pressurization task is common to the propulsion subsystem where it is discussed in greater detail.

4. **Pioneer Environment Control**

This task is that of furnishing a satisfactory environment to each of the reconnaissance components. The firm requirements on the electronic equipment will be furnished by the results of the environmental simulation tests and verified in the STV tests. The required pressurization can be furnished by the pressure system and the operating temperature range can be maintained at whatever is required at or slightly above room temperature. The techniques of temperature control are based on the work reported in the appendix.

5. **Advanced Airframe Structure and Mechanisms**

The airframe of the Advanced system will accommodate a larger propulsion system and establish a larger payload on orbit. This design must be based on the actual performance achieved with the Atlas. The vehicle design will not change basically from that reported in Ref. a, the principal changes being in propellant tank size, and possible system for mechanically positioning the reconnaissance system components, such as cameras and antennae, and unfolding those antennae whose sizes have required folding during the boost phase. Such configurations as that of the Bola, (Ref. b) will require further analysis and must await the results of tests which may be done during the early o[illegible] achievement.

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R&D PROJECT CARD
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MSD 1536

1. PROJECT TITLE SUBSYSTEM DEVELOPMENT PLAN - AIRFRAME FOR ADVANCED RECONNAISSANCE SYSTEM (UNCLASSIFIED) (PIED PIPER)	2. SECURITY OF PROJECT ~~Secret~~	3. PROJECT NUMBER 1115
	4.	5. REPORT DATE 1 March 1956

6. **Advanced Booster Modification and Separation System**

For the Advanced system the payload capability of the booster must be increased. At present it appears that the optimum gross weight of the orbiting vehicle will be about 14,000 pounds. This will require strengthening the front end of the booster. This modification will include the separation system.

7. **Advanced Pressure System**

Although designs have not been worked out specifically for the Advanced vehicle, it is anticipated that the pressurization system would be essentially as indicated for the Pioneer system including the capability of pressure stabilizing the tanks. It may be noted that a reduction of over 60% in pressurization system weight could be achieved if liquid oxygen is available in the missile for refrigerating the helium.

8. **Advanced Environment Control**

This task will become firm only after observations are made of the operation of the Pioneer system. The temperature control will be based on extensions of the work reported in the appendix.

9. **Bola and Other Special Configurations**

This task will be defined when the various nuclear and solar power units are specified. These special configurations must furnish some method of reducing the nuclear radiation field in which the reconnaissance equipment operates. The Bola configuration, (Ref. b) which separates the radiation source from the reconnaissance gear, will be examined in greater detail. The design configurations capable of incorporating a solar power unit must await the collecting of further information concerning the orbital environment. The configuration incorporating the infrared warning system will be made after the development of the detection and communication equipment, and a determination of the best altitude of the orbit.

21 d. Not Applicable

21 e. Not Applicable

21 f. Not Applicable

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R&D PROJECT CARD
CONTINUATION SHEET

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MSD 1536

1. PROJECT TITLE SUBSYSTEM DEVELOPMENT PLAN - AIRFRAME FOR ADVANCED RECONNAISSANCE SYSTEM (UNCLASSIFIED) (PIED PIPER)	2. SECURITY OF PROJECT ~~Secret~~	3. PROJECT NUMBER 1115
	4.	5. REPORT DATE 1 March 1956

21 g. **References**

a) First Quarterly Progress Report, Pied Piper, Vol. IV, MSD 1363, 1 November 1955 (S)

b) Bola Configuration, Task 16, Pied Piper Progress Report for November 1955, MSD 1440 (S)

21 h. Not Applicable

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Pied Piper

MSD 1536

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MISSILE SYSTEMS DIVISION

TABS

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Subsystem A - AIRFRAME

Tab I - General Design Specifications

I. GENERAL

A. Statement of the Problem

An airframe must be designed which will:

1. Permit achievement of maximum reconnaissance capability
2. Be compatible with the WS 107 system
3. Permit achievement of orbit capability by the end of 1958

B. Approach

A vehicle will be designed which will meet the requirements of (2) and (3) above, fulfilling insofar as possible the maximum reconnaissance capability. The constraints imposed by (2) and (3) establish the size and weight and dictate the choice of certain components. This will be the "Pioneer Vehicle."

To more adequately comply with the requirements of (1), an improved reconnaissance vehicle will be designed (called "Advanced Vehicle") which will utilize the performance capabilities of the Atlas booster but will assume relaxation of the physical and temporal limitations. The advanced vehicle will provide a possible orbiting capability, but at reduced payload, using boosters of lower performance than Atlas such as IRBM or large, solid rockets.

C. Solution and Recommendation

1. Airframe Configuration

The configurations of the Pioneer and Advanced vehicles are basically the same - a cylindrical body with a conical nose. The diameter of the cylinder of the Pioneer is determined by the available diameter of the Atlas booster, 61 inches. The length of the Pioneer vehicle is 18 feet, which is the maximum allowed by the height of the present launching stand. The dimensions of the Advanced vehicle will be made compatible with the modified Atlas booster and the requirements of the propulsion and other basic subsystems as well as with the larger payloads. Our examinations have shown that the shape of the Pioneer vehicle has very little effect on the performance of the Atlas booster. The shape will be determined therefore, by considerations of aerodynamic heating in the early boost phase and by equipment packaging requirements. Investigation has shown that for the given shape the skin thicknesses shown in the appendix were necessary. The skin of the Pioneer vehicle furnishes protection for the components during the boost phase and temperature control for the various reconnaissance components during the orbit phase. In this connection it should be noted that the skin and structure aft of the tank section is in reality an adapter section which remains with the booster upon separation of the orbital unit.

Drawings of the configuration and the inboard profile are included in the appendix.

2. Inboard Profile

The principal components in the center section of the airframe are the propellant and pressurization gas tanks. These are basically spherical in shape and are nested so that each tank is partially submerged in the next larger tank. The high pressures required of a gas-fed system dictated the shape of the tank and for these shapes the nested configuration of the tanks was chosen from consideration of the resulting strength and rigidity.

The configuration chosen for the Pioneer vehicle is easily produced. The engine is mounted from the bottom of the most rearward tank and is the largest component in the aft section of the vehicle. In addition to the two control engines mounted on opposite sides of the main engine nozzle, several antennas and payload components can be mounted in the annulus surrounding the engine. These components are exposed upon separation of the satellite vehicle from the sustainer, because the entire adapter section remains with the Atlas sustainer. The components can be opened out and reoriented as necessary during orbiting.

The conical nose compartment contains the auxiliary power source, the guidance and control unit, and other small payload components as required. It should be noted that an ogival nose shape can be accommodated which will allow more space for payload components, if needed.

Communication and power transmission between the forward and after compartments can be accomplished through the space between the tanks and the outer skin. This space also contains the pressurization system lines and valves, and provides room for the mounting of slot antennas in the outer skin.

The profile of the Advanced vehicle has not been chosen. However, the requirement for larger quantities of propellant will permit a tank structure which is integral with the skin which will give a completely pressure-stabilized system, achieving some saving in weight.

3. Structure

The tank assembly of the Pioneer vehicle is the structural backbone of the vehicle. Thrust loads from the booster are transmitted through the structure of the adapter section to the conical support for the propellant tanks. The adapter section is in effect a circular column consisting of a magnesium skin stiffened by several equally spaced longerons and circumferential rings. The boost loads are transmitted as tensile stresses in the tank walls to the engine mount. Conversely, when the engine is thrusting, its loads are transmitted through the engine mount to the tank bottoms and thence, like the booster loads, they are transmitted via the pressurized gas to the tank head which supports the auxiliary power source, guidance and control package, and other payload units. Thrust is transmitted to the airframe through another conical structural unit from the forward tank to the base of the nose cone. The nose cone is itself only an aerodynamic shield for the forward part of the missile. The region of external skin between the two thrust cones is entirely non-structural and is used for mounting slot antennas.

4. Compatibility with Atlas

The achievement of compatibility with the present Atlas system has determined the design of the Pioneer vehicle. This vehicle will be carried by the booster using an adapter section which will fit the attachment rings for the present nose. The adapter section will remain with the booster on separation. The length of the vehicle is determined by the clearance between the Atlas booster and the gantry crane of the present launcher. The weight of the early versions of the Pioneer vehicle will be that of the present Atlas booster payload; this will be increased to the maximum safe booster loading as the program develops. The only changes in the present Atlas system will be in the trajectory.

The Advanced vehicle will require modification of the Atlas system to utilize the complete capability of the Atlas booster, which will involve redesign of the launcher and the front of the booster.

5. Aerodynamics

Investigation of the aerodynamics of the Atlas vehicle indicates that about 2.5 per cent of the total impulse of the propulsion system is dissipated in the drag effect. Further, it is indicated that redesigning the Pioneer vehicle can only change this between the limits of 2.4 to 2.6 per cent, which indicates that some other requirement will dictate the size and shape. For example, the aerodynamic heating during the early stages of the boost is more important. The Pioneer vehicle is designed so that its temperature rise will not degrade its structural integrity.

6. Performance

The performance of the Atlas was investigated for the present payload of 3,500 pounds and for the present maximum payload capability of 7,000 pounds. The Pioneer vehicle was designed to use this performance and to give the additional performance required to achieve orbital capability with payloads adequate to accomplish visual reconnaissance. The Advanced vehicle is designed to achieve the maximum payload on orbit for the performance of the Atlas booster. The complete picture of the performance is shown in the appendix.

7. Environmental Control

The airframe will furnish temperature control for the payload components. Present studies indicate that using the power dissipated together with properly designed insulation and radiation shields the correct operating temperatures for the components can be maintained. The design will be made after tests in the environmental chamber have been completed. No other type of environmental control will be performed by the airframe.

8. G.F.E.

The overall environmental requirements indicate that no part of this subsystem will be G.F.E.

9. Destruct System

A destruct system will be incorporated in the orbiting vehicle. This system will destroy the capability of the vehicle to reenter the atmosphere successfully. It does not seem necessary to employ a destruct system which will operate during the boost phase since the Atlas has such a system.

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Subsystem A - Airframe

Tab 2 Summary - Subsystem Milestones

System Study and Design Begins

Pioneer Vehicle Mockup

Release of Drawings for Development Pioneer Vehicle

Fabrication of Development Pioneer Vehicle

Testing of Development Pioneer Vehicle

Engineering Design Freeze for Pioneer

Release Eng. Dwgs. to Shop

Pioneer Production Vehicle

Ground Firing of First Tied down OTV

Testing of First Production Vehicle at STF

Flight of First NOTV

Flight of First OTV

Advanced Booster Modification Mockup

Advanced Vehicle Mockup

Release of Drawings for Development Advanced Vehicle

Testing of Development Advanced Vehicle

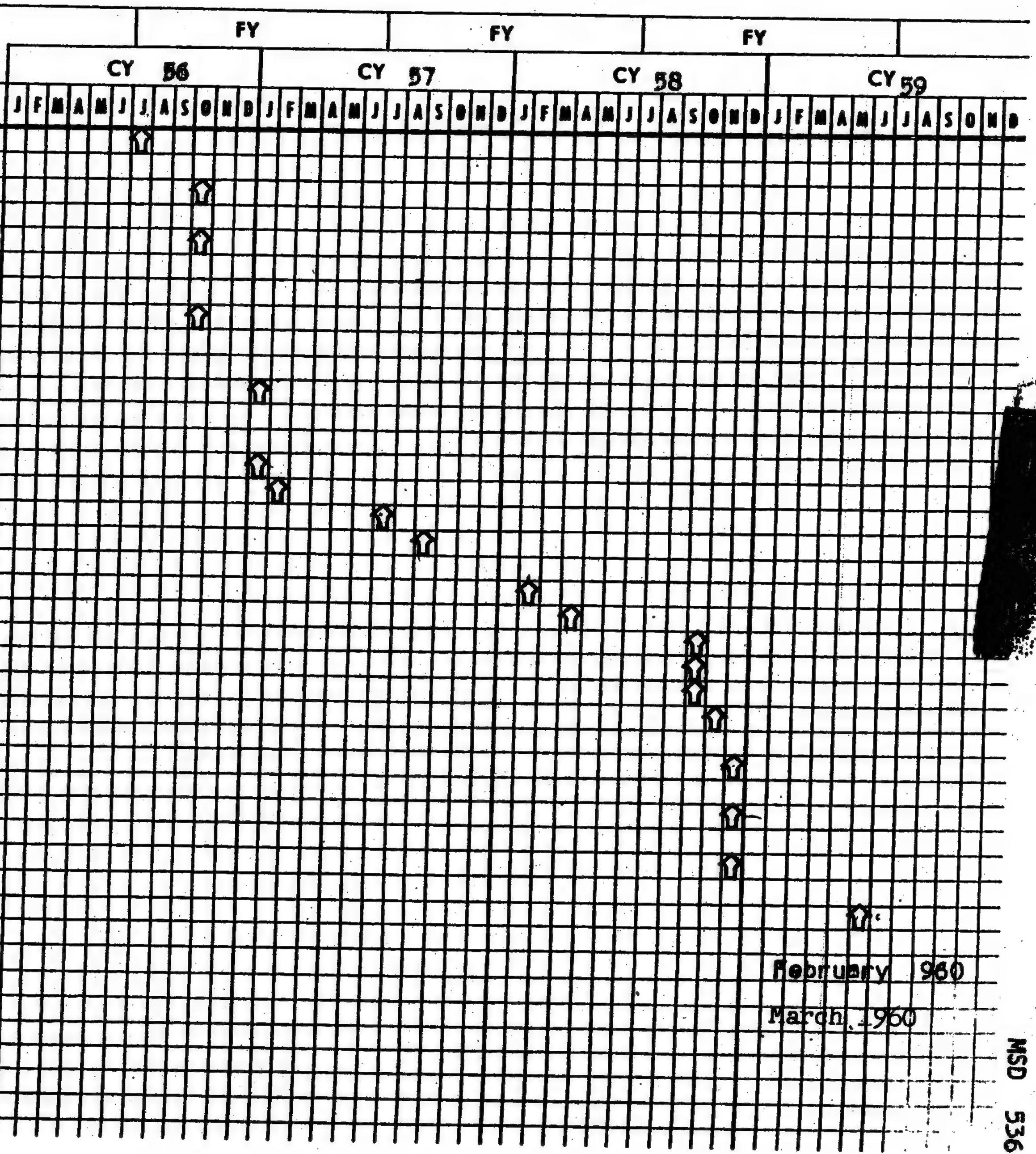
Engineering Design Freeze of Advanced Vehicle

Release Eng. Dwgs. for Advanced Production Vehicle

Ground Firing of First Advanced Production Vehicle

Testing of First Advanced Production Vehicle at STF

Flight of First Advanced Production Vehicle at STF



February 1960

March 1960

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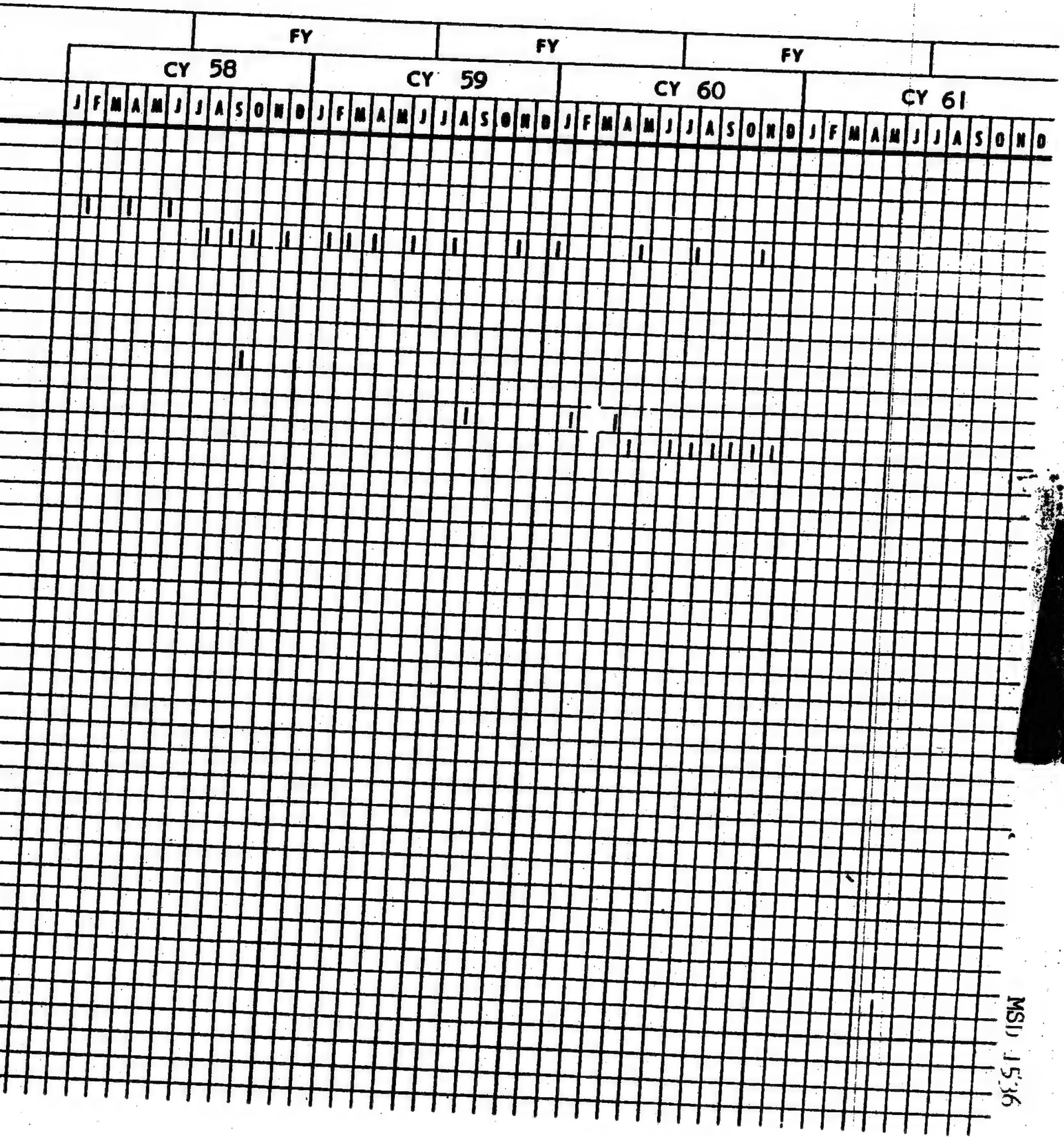
Subsystem A - Airframe

Tab 2 Summary - Hardware Delivery

Pioneer Airframe
Mockup (Oct/56)
Development Vehicle
Production Vehicle

Advanced Airframe
Mockup
Development Vehicle
Production Vehicle



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LOCKHEED AIRCRAFT CORPORATION

Subsystem A - Airframe

Tab 2 Summary - Subsystem Tests

								FY												FY												FY																	
		CY 56												CY 57												CY 58												CY 59											
		J	F	M	A	M	J	J	A	S	O	N	D	J	F	M	A	M	J	J	A	S	O	N	D	J	F	M	A	M	J	J	A	S	O	N	D	J	F	M	A	M	J	J	A	S	O	N	D
1	Research Lab Tests:																																																
2	a. System compatibility tests in																																																
3	mockup										■																																						
4	b. Leak tests of tanks and																																																
5	compartments													■																																			
6	c. Lab test of separation system													■																																			
7																																																	
8																																																	
9	Tests of Temperature Control Designs																																																
10	in Environmental Control Facility													■																																			
11																																																	
12																																																	
13	Flight tests of temperature control																																																
14	systems in STV																		■																														
15																																																	
16																																																	
17	Flight tests of separation systems																																																
18	in STV																		■																														
19																																																	
20																																																	
21	Pioneer Airframe																																																
22	a. Environmental Tests																		■																														
23	b. Ground Firing Performance Tests																							■																									
24																																																	
25																																																	
26	Flight tests of Temp. Control Systems																										■																						
27	in OTV																																																
28																																																	
29																																																	
30	Advanced Airframe																																																
31	a. Environment Tests																																					■											
32	b. Ground Firing Performance Tests																																					■											
33																																																	
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35																																																	
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37	First Tests Shown																																																
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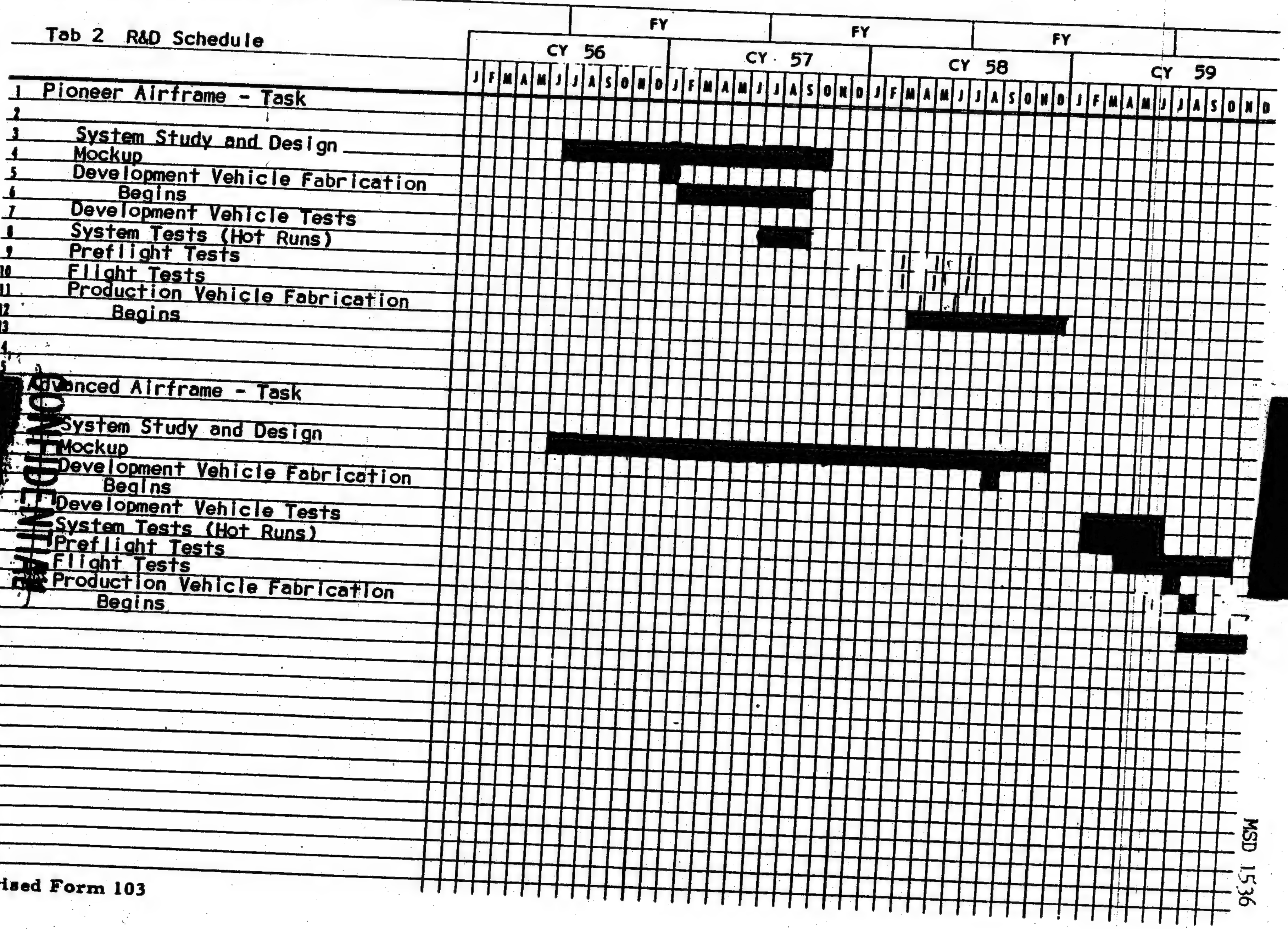
Subsystem A - Airframe

Tab 2 R&D Schedule



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MSD 1536

1.

R & D TEST ANNEX

☐ SYSTEM ☒ PROJECT ☐ TASK ☐ OTHER

2. REPORTS CONTROL SYMBOL

PAGE 1 OF 1 PAGES

3. DATE 1 March 1956

4. TITLE Subsystem A - AIRFRAME

5. INITIAL ☐ CHANGE

6. NUMBER

7. RESP CENTER

8. PROJECT OFFICER

9. SUPPORTS (Sys or Proj)

10. CONTRACTOR

11. CONTR NR

12. PRIORITY AND PREC

13. SECURITY

14. ITEM NUMBER	15. TEST ITEM	16. TEST DESCRIPTION	17. TEST AGENCY AND SITE	18. TEST ITEM AVAILABLE	19. RQD TEST COMPL DATE
1	Airframe-booster Configuration	High Speed Aerodynamic Tests	NACA, Ames	Oct 1956	Feb 1957
2	Simulated Pioneer Airframe Configuration	System Test Vehicle Simulated flight tests	AFMTC - GFF	Oct 1956	Continui [redacted]
3	Full Scale Pioneer Airframe	Captive flight test	LMSD - STF	Sep 1957	Oct 195 [redacted]
4	Airframe-Booster Configuration	Non orbiting flight test	AFMTC - GFF	Apr 1958	Aug 19 [redacted]
5	Simulated Advanced Airframe Configuration	System Test Vehicle flight test	AFMTC - GFF	Sep 1959	Feb 19 [redacted]
6	Full Scale Advanced Airframe	Captive Flight Test	LMSD - STF	Oct 1959	Nov 1959

TEST CENTER APPROVAL

20. NAME ORGANIZATION DATE

21. NAME ORGANIZATION DATE

RESPONSIBLE CENTER APPROVAL

22. NAME ORGANIZATION DATE

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MSD 1536

1.

R & D TEST AND TEST SUPPORT AIRCRAFT ANNEX

☐ SYSTEM ☒ PROJECT ☐ TASK ☐ OTHER

2. *REPORTS CONTROL SYMBOL*

PAGE 1 OF 1 PAGES

3. DATE 1 March 1956

4. TITLE Subsystem A - AIRFRAME

5. INITIAL ☐ CHANGE

6. NUMBER

7. ITEM NUMBER	8. AIRCRAFT REQUIRED			9. ASG CODE	10. MOD REQD	11. DATE REQD AND LOCATION	12. ESTIMATED RELEASE DATE	13. RECOMMENDED DISPOSITION	14. F HRS F CST	15. EST COST
	QTY	TYPE, MODEL AND SERIES	SERIAL NUMBER							

AIRCRAFT WILL NOT BE REQUIRED FOR TEST OF THE AIRFRAME SUBSYSTEM.

ARDC FORM 1 JUL 55 106 PREVIOUS EDITIONS OF [illegible]

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[redacted]

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[redacted]

MSD 1536

1.

R & D MATERIEL ANNEX

☐ SYSTEM ☒ PROJECT ☐ TASK ☐ OTHER

2. REPORTS CONTROL SYMBOL

PAGE 1 OF 1 PAGES

3. DATE 1 March 1956

4. TITLE Subsystem A - AIRFRAME

5. INITIAL ☐ CHANGE

6. NUMBER

7. MATERIEL REQUIREMENTS *(Indicate Items in Columnar Form using Columns as cited in Examples)*

Specialized material or equipment is required for this subsystem that cannot be deliniated at this time.

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[redacted]

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[redacted]

SUBSYSTEM A - AIRFRAME SUBSYSTEM

DATE 1 March 1956

LOCATION:

ITEM: SYSTEM TEST FACILITY *
AFMTC *

BUDGET CONTROL ESTIMATE:

USING AGENCY: LOCKHEED - MSD

NEED DATE:

SCHEDULE:	1956						1957												1958												1959											
/	J	A	S	O	N	D	J	F	M	A	M	J	J	A	S	O	N	D	J	F	M	A	M	J	J	A	S	O	N	D	J	F	M	A	M	J	J	A	S	O	N	D

DESCRIPTION AND UTILIZATION:

* Complete descriptions of these facilities are given in Tab 6, Subsystem L - Vehicle Ground Support

REMARKS:

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Tab 7

R & D Contract Funds

Subsystem A - Airframe

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MSD 1536

Subsystem A. AIRFRAME

Tab 7. R & D Contract Funds (in thousands of dollars)

	FY 57				FY 58				FY 59				FY 60	
	1	2	3	4	5	6	7	8	9	10	11	12	13	14
LAC														
(1) Research and Engineering	86	147	199	289	347	368	408	423	398	382	344	344	214	239
(a) Sub Contracts														
(2) Fabrication	490	490	504	858	862	971	1,222	1,367	1,364	1,020	977	977	751	1,139
(b) Purchased Components														
Sub Total	577	638	704	1,148	1,209	1,340	1,630	1,791	1,762	1,403	1,321	1,321	965	1,378
Fee	57	63	70	114	120	134	163	179	176	140	132	132	96	137
TOTAL*	635	702	775	1,263	1,331	1,470	1,794	1,970	1,938	1,543	1,454	1,454	1,063	1,517
Total Fiscal Year				3,375				6,565				6,389		
* Differences in totals due to rounding														

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Subsystem A. AIRFRAME

Tab 7. (Cont) R & D Contract Funds (in thousands of dollars)

	FY 60		FY 61				FY 62				FY 63				Totals*
	15	16	17	18	19	20	21	22	23	24	25	26	27		
LAC															
(1) Research and Engineering	294	314	324	314	309	294	254	244	261	236	241	226	147		7,661
(a) Sub Contracts															
(2) Fabrication	990	1,667	1,862	1,604	1,674	1,827	1,331	1,476	1,360	1,942	2,500	589	-0-		31,826
(b) Purchased Components															
Sub Total	1,284	1,982	2,186	1,919	1,984	2,121	1,586	1,721	1,622	2,179	2,742	815	147		39,487
Fee	128	198	218	191	198	212	158	172	162	217	274	81	14		3,948
TOTAL*	1,414	2,181	2,406	2,112	2,183	2,334	1,745	1,894	1,784	2,397	3,017	898	162		43,436
Total Fiscal Year		6,175				9,035				7,820		4,077			43,436

* Differences in totals due to rounding

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Tab 8

Estimate of Manpower Requirements

Subsystem A - Airframe

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Subsystem A AIRFRAME

Tab 8 Estimate of Manpower Requirements

WORK ITEM	Type of Manpower	QUARTERS													
		1	2	3	4	5	6	7	8	9	10	11	12	13	14
LAC Research and Development	1-2-3 *	17	29	38	55	66	70	80	83	78	75	69	69	43	48
LAC Fabrication and Assembly	4	138	138	138	235	236	266	345	386	385	288	288	288	217	329
TOTAL		155	167	176	290	302	336	425	469	463	363	357	357	260	377
	1														
	2														
	4														
*Average:															
40% Type 1 - Scientific and Technical															
50% Type 2 - Engineering Support															
10% Type 3 - Management and Administration															

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Subsystem A AIRFRAME

Tab 8 Estimate of Manpower Requirements (Cont.)

WORK ITEM	Type of Manpower	QUARTERS														TOTAL MAN QUARTERS
		15	16	17	18	19	20	21	22	23	24	25	26	27	28	
LAC Research and Development	1-2-3 *	59	63	65	63	62	59	51	49	52	47	48	45	29		1512
LAC Fabrication and Assembly	4	285	480	536	462	483	527	384	426	390	557	717	169	-0-		9093
TOTAL		344	543	601	525	545	586	435	475	442	604	765	214	29		10605
	1															
	2															
	4															
*Average:																
40% Type 1 - Scientific and Technical																
50% Type 2 - Engineering Support																
10% Type 3 - Management and Administration																

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Pied Piper

MSD 1536

LOCKHEED AIRCRAFT CORPORATION
MISSILE SYSTEMS DIVISION

APPENDIX

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Subsystem A, AIRFRAME

APPENDIX

1. AIRFRAME

The intent of this section is to present a vehicle system design for the ARS. The following study refers primarily to vehicles of the Pioneer order.

The basic problem requires:

1. Vehicle orbital capability with an acceptable payload.
2. Development of equipments and systems which would be compatible with an orbit environment.

1.1 Approach

Time-scale requirements are the predominating influence on early vehicle systems. The following ground rules were adopted as the most promising for achievement of the desired end product.

1. The evolution of system should provide a high probability of achieving an artificial satellite in time for the IGY.
2. The XSM-65 Series "C" Convair Atlas missile is to be used as the booster unit for the orbiting stage unit. The orbiting stage unit replaces the Atlas warhead (3500 pounds). The orbiting stage must be designed to be compatible with the ground handling and launching systems equipment for the Atlas missile.

3. The orbiting stage propellant tankage and propellant feed systems are to accommodate the amount of propellant required if the Atlas booster range capability is degraded by 30%.
4. The orbiting stage propulsion system is to be a liquid propellant rocket type in the 5,000 to 10,000 lb thrust (in a vacuum) category.
5. The function of stage separation for the satellite vehicle will occur shortly after the booster burn-out to enable the orbiting vehicle attitude control system to provide attitude control of the orbiting vehicle during the long coasting period in the trajectory, and prior to starting of engines of the orbital vehicle.
6. The aerial reconnaissance system capability potential will be demonstrated using two separate versions of the same basic vehicle.
7. The payload is defined as only those equipments or energy sources which are carried aloft for performing functions necessary after the machine is established in an orbit.

The ground rules adopted suggest the direction of approach and in some cases establish design philosophy.

1.2 Design Concepts

The first four ground rules have the major influence on the satellite design. The vehicle system that has evolved incorporates no major deviations from the present state of the art and the limited development necessary presents no unusual technical problems. The vehicle configuration and system designs are not completely optimized in view of the adopted ground rules. However, comparison studies were made involving major vehicle structure, and the effects of varying propulsion systems, to back up the vehicle configuration presented.

The first ground rule reduces the hardware selection problem to a matter of availability. Propulsion possibilities explored, including engines proposed or under development by several engine manufacturers, ranged in thrust from 7500 to 9500 lb in a vacuum.

Designs using many engines and various propellant combinations were considered. The power plant selected was the Aerojet Vanguard gas pressure-fed engine with a thrust in a vacuum of 7500 lb, a specific impulse of 278 sec, and utilizes UDMH + WFNA as propellants.

The second ground rule limits the size and design weight of the orbiting vehicle, and it also imposes the major structural design criteria for all structure of the orbiting vehicle with the exception of the tank system. The length of the configuration (216 inches) was determined by the consideration of gantry clearance of the SM-65 Atlas in its launching preparation and servicing equipment. This length clears the gantry structure by approximately three feet, which will permit a chain hoist to traverse the gantry and also clear the nose section of the orbital vehicle when it is in place on the Atlas. The orbiting stage vehicle diameter was held to 61 inches, which permits its installation on the booster with no structural rework of the booster.

Model trajectories using the SM-65 Atlas booster were calculated, which indicated the system boost phase or exit physics design requirements. Flight path tangential accelerations were plotted versus altitude and versus time; also Mach number vs. altitude was plotted. This information permitted the determination of basic information for structural and system design requirements such as:

1. Acceleration loads on orbiting stage structure, systems, and components.
2. Aerodynamic structural heating effects.
3. Structural system dynamics investigations.

The third ground rule was adopted on the basis of insurance for achieving orbiting even if booster performance is 30% below par. It is logical to assume such booster degradation in the early firing phases of the Atlas program but the figure is probably 30% pessimistic. The tankage used in the Pioneer vehicle is approximately three times greater than would be necessary if no booster range degradation were considered. This extra tankage is especially significant in view of the selection of a gas pressure fed propulsion system which requires high tank pressures of the order of 300 psi.

Ground rule 4 limits the use of solid rockets. In this connection, there are problems of obtaining acceptable specific impulse, vehicle path control, and engine thrust cut-off.

1.3 Solution and Recommendations

Vehicle sizes andgeneral arrangement are shown in Figs 1-1 through 1-4. The arrangements for the basic OTV, Pioneer Visual Reconnaissance, and Pioneer Ferret versions are included. The photo reconnaissance and ferret versions are accomplished by adding their respective equipments and supporting structural systems to the basic OTV.

During the boost Atlas phase of the OTV trajectory, a tangential acceleration of 6.5 g's is experienced with a payload of 3500 lb. For design purposes a figure of 8.0 g's was used in anticipation of the possible use of a wide range of trajectories. A transverse or lateral loading of 1.0 g was employed. The aerodynamic heating experienced by the structure permitted the use of 0.020 stainless steel or 0.070 magnesium alloy at a point two feet aft of the end of the nose cone. The design tangential acceleration during sustainer engine operation of the OTV was 5 g's.

The vehicle consists of a working hard core missile system covered by a thin semi-non-working shell. The working hard core is the load transfer and load supporting agent to which is attached all equipments and subsystems and consists of three nested pressurized spherical tanks, having the rocket engine attached at one end of the combination, and the guidance and control package and orbiting stage auxiliary power system attached to the other end.

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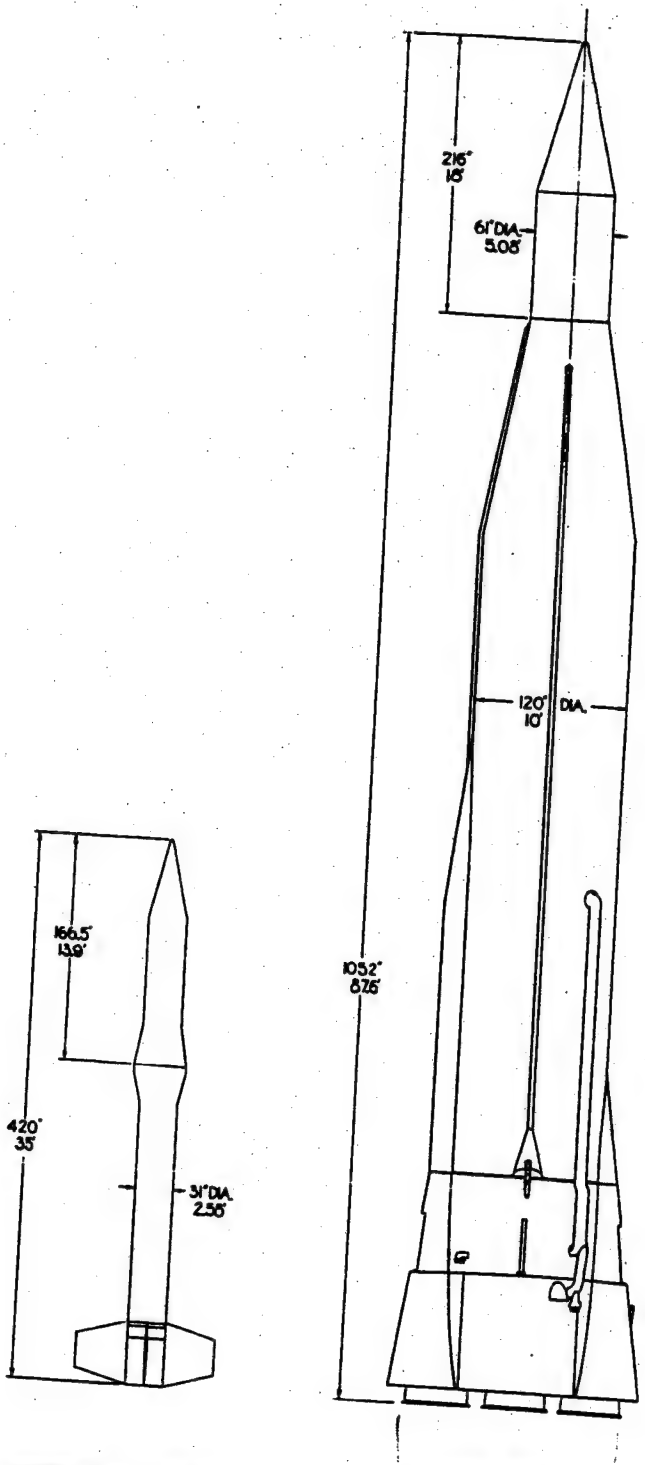

SYSTEMS TEST VEHICLE

SATELLITE VEHICLE
WITH
ATLAS BOOSTER

Fig. 1-1 Systems Test Vehicle Compared with Satellite Vehicle

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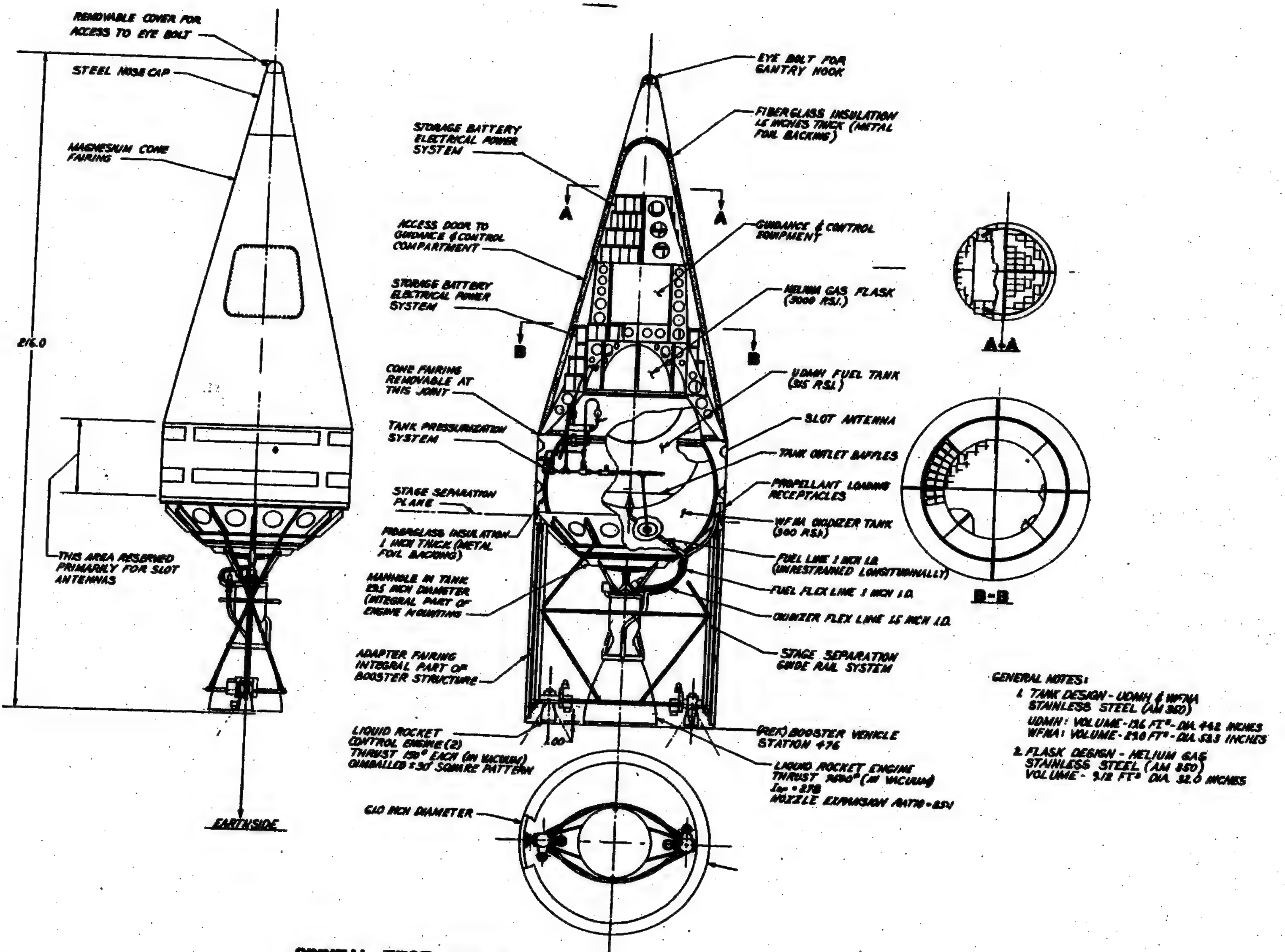
REMOVABLE COVER FOR ACCESS TO EYE BOLT
STEEL NOSE CAP
MAGNESIUM CONE FAIRING
216.0
THIS AREA RESERVED PRIMARILY FOR SLOT ANTENNAS
EARTHSIDE
STORAGE BATTERY ELECTRICAL POWER SYSTEM
ACCESS DOOR TO GUIDANCE & CONTROL COMPARTMENT
STORAGE BATTERY ELECTRICAL POWER SYSTEM
CONE FAIRING REMOVABLE AT THIS JOINT
TANK PRESSURIZATION SYSTEM
STAGE SEPARATION PLANE
FIBERGLASS INSULATION 1 INCH THICK (METAL FOIL BACKING)
MANHOLE IN TANK 23.5 INCH DIAMETER (INTEGRAL PART OF ENGINE MOUNTING
ADAPTER FAIRING INTEGRAL PART OF BOOSTER STRUCTURE
LIQUID ROCKET CONTROL ENGINE (2) THRUST 150# EACH (IN VACUUM) GIMBALLED ±30° SQUARE PATTERN
61.0 INCH DIAMETER
1.00
A
A
B
B
EYE BOLT FOR GANTRY HOOK
FIBERGLASS INSULATION 1.5 INCHES THICK (METAL FOIL BACKING)
GUIDANCE & CONTROL EQUIPMENT
HELIUM GAS FLASK (3000 P.S.I.)
UDMH FUEL TANK (315 P.S.I.)
SLOT ANTENNA
TANK OUTLET BAFFLES
PROPELLANT LOADING RECEPTACLES
WFNA OXIDIZER TANK (300 P.S.I.)
FUEL LINE 1 INCH I.D. (UNRESTRAINED LONGITUDINALLY)
FUEL FLEX LINE 1 INCH I.D.
OXIDIZER FLEX LINE 1.5 INCH I.D.
STAGE SEPARATION GUIDE RAIL SYSTEM
(REF.) BOOSTER VEHICLE STATION 476
LIQUID ROCKET ENGINE THRUST 3400# (IN VACUUM) Isp = 278 NOZZLE EXPANSION RATIO = 25:1
A-A
B-B

GENERAL NOTES:

1. TANK DESIGN - UDMH & WFNA STAINLESS STEEL (AM 350)
   UDMH: VOLUME - 12.6 FT³ - DIA 44.2 INCHES
   WFNA: VOLUME - 23.0 FT³ - DIA 53.3 INCHES
2. FLASK DESIGN - HELIUM GAS STAINLESS STEEL (AM 350) VOLUME - 9.12 FT³ DIA 32.0 INCHES

ORBITAL TEST VEHICLE

Fig. 1-2 Orbital Test Vehicle, Inboard Profile

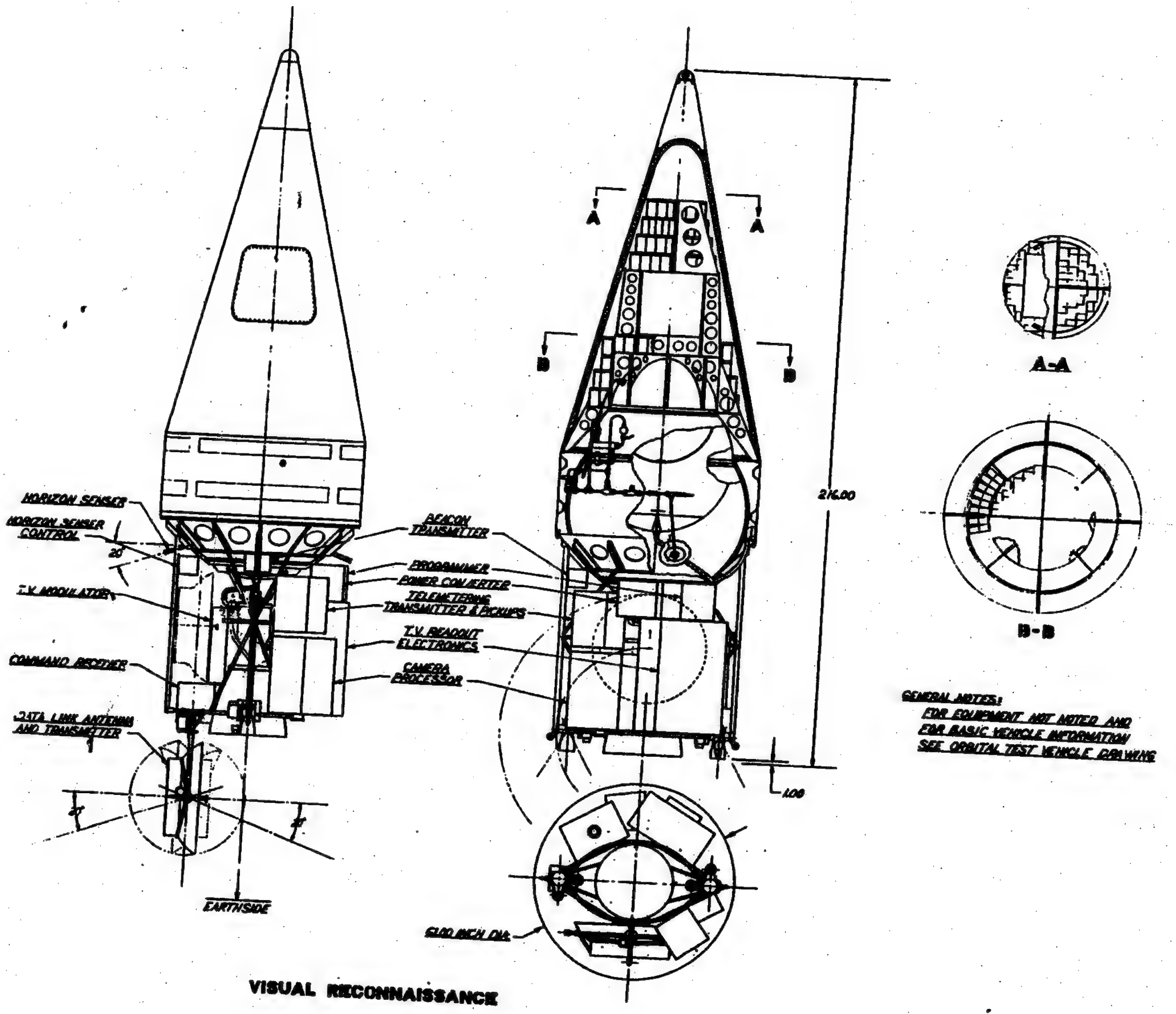

Fig. 1-3 Visual Reconnaissance, Inboard Profile

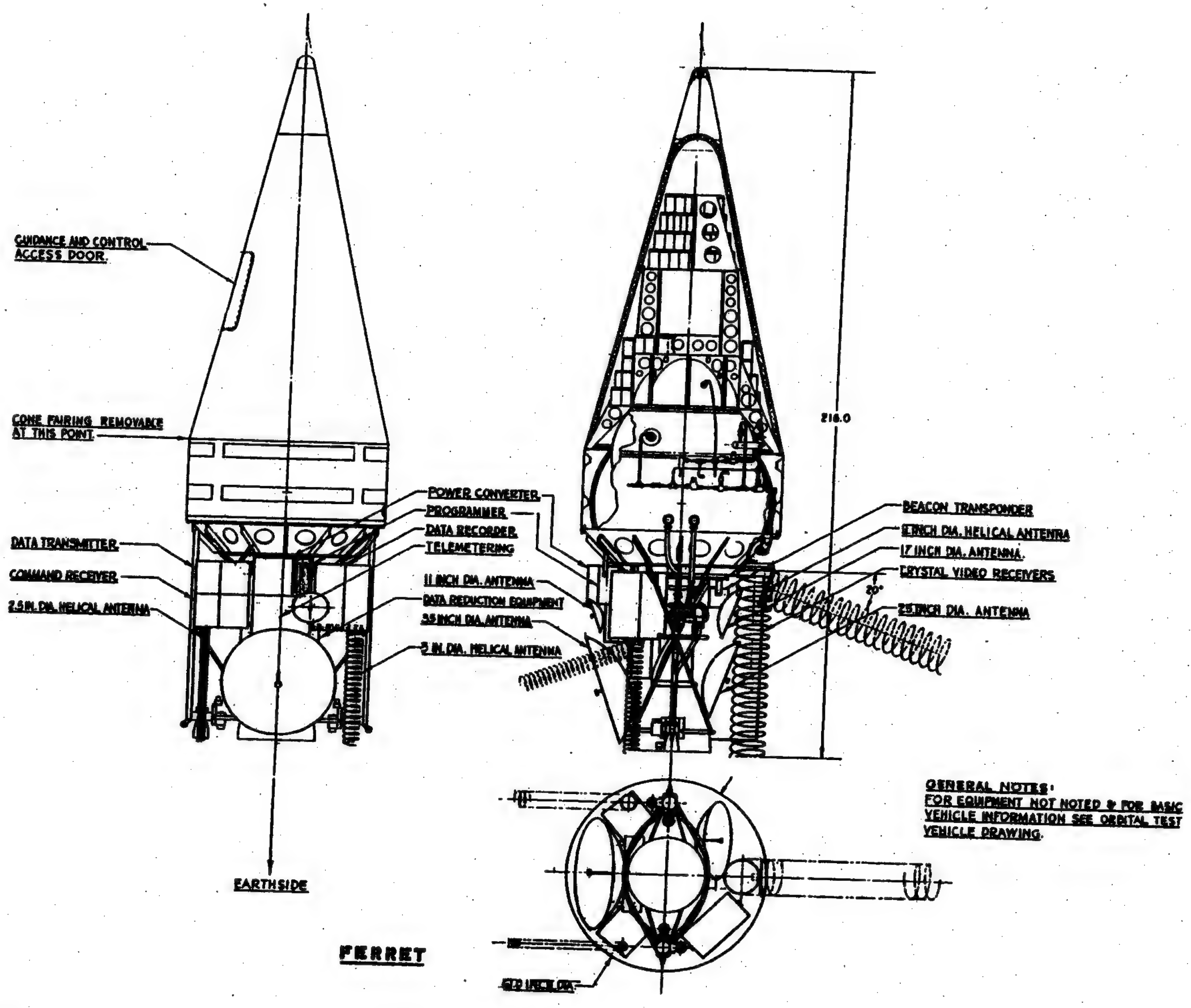

Fig. 1-4 Ferret, Inboard Profile

The tank assembly upper sphere is a 3000 psi helium gas storage vessel which supplies the working pressure for the propellant feed system. The top surface of the storage vessel incorporates a structural rack and supporting system for the guidance and control systems, and the orbiting stage storage battery power source. A chemical APU system with appropriate APU propellant tankage could be installed in place of the battery system without undue difficulty. The center tank of the tank assembly contains the UDMH at an operating pressure of 315 psi and the lower tank is a container for the WFNA at an operating pressure of 300 psi. The lower end of the oxidizer tank is a part of the engine mounting system which bolts in place to close out the tank.

The Ryan Aeronautical Company was consulted on the tank design from a manufacturing feasibility and material selection (AM 350) point of view. Ryan technical people and sales representatives made it known that the tank could be built by Ryan or others without undue difficulty and within the time-scales required.

The rocket propulsion system shown is a version of the Aerojet Vanguard gimbaled engine. A 20:1 nozzle expansion ratio is used. The two small gimbaled rocket control engines, each with a 150 lb thrust in a vacuum, are added, and operate from the main engine system.

The main thrust chamber shown is fixed and all powered flight control of the orbiting vehicle is achieved by the small control engines. Study may indicate that the Vanguard engine can be used without modification, if roll control can be accomplished using the orbiting attitude control system during powered flight, or possibly by using gas jets in combination with the attitude control system. If the gimbaled engine is used, a one foot increase in vehicle length will probably result, and some oxidizer tank revision will have to be made, since the Navy engine incorporates a conical engine mount that also serves as a tank bottom.

The first two feet of the nose cone fairing is of stainless steel and incorporates a lifting eye for hoisting the vehicle. The lifting eye is covered by a removable steel cap. The remainder of the ring conical fairing is 0.070 magnesium alloy. Access to the guidance and control compartment is provided. The complete conical fairing is removable. The nose compartment is thermo-insulated with fiberglas insulation backed with a radiant heat foil reflector. The compartment can be pressurized if necessary.

The short cylindrical section fairing is reserved for various slot type antenna installations. The antenna dielectric will be of sintered alumina or some other heat-resistant dielectric.

The orbiting vehicle is attached to the Atlas booster by means of a cylindrical adapter or skirt section. The adapter is permanently fixed by bolting (or other means) to the Atlas at approximately vehicle station 425. It is a magnesium alloy semi-monocoque structure utilizing skin, longitudinal members, and frames. The forward end of the adapter is connected to orbiting vehicle utilizing a machined ring and clamp arrangement. The longitudinal thrusting forces during the boost phase of the trajectory, are transferred by the adapter section to the orbital stage. The loads are transferred from the adapter to the orbiting vehicle tank structure by a truncated cone sheet tension member. The adapter section is capable of handling the loads imposed on it without benefit of pressure stabilization of the shell. This alleviates the pressure seal problem somewhat, and will permit greater ease in handling. It will also permit the inclusion of doors for access to the orbiting vehicle engine and equipment compartment without undue difficulty.

Staging separation of the orbital vehicle occurs shortly after burn-out of the booster stage at an altitude of approximately 715,000 ft. The effects of atmosphere on the system are negligible at this altitude. When the signal is received for separation, an electric impulse will ignite an explosive charge, severing a mechanical connection of the adapter clamping ring which is in a state of tension. The ring will spring outward and away from the separation joint and the two bodies will be free to separate. The compartment between the orbiting stage and booster will contain air at sea level atmospheric pressure. The air under pressure will provide the necessary impulse to cause a velocity differential between the two bodies. At the end of the long coasting period the orbital vehicle engines will be started. The separation act will have the benefit of control afforded by the orbital vehicle attitude control system. The separation takes place essentially in a gravitation-free field. For additional reliability a track and roller guide system has been incorporated into the design to insure the proper emergence of the overhanging equipments on the orbiting vehicle. Should further investigation reveal the need for additional impulse to cause the body velocity differential at separation, a simple mechanical system can be incorporated.

The in-flight pressurization system for the Pioneer Reconnaissance Vehicle is a simple helium system primarily for the purpose of providing pressure to the main propellant tanks. The principal reason for pressurizing the propellant tanks is to drive the propellant constituents into the combustion chamber at pressures sufficiently high to make turbopumps unnecessary. Furthermore, the internal tank pressures provide rigidity and load carrying capability to the propellant tanks. The pressurization system can also provide gas to any payload component that may require a pressurized environment.

The pressurization system consists primarily of a spherical high pressure storage tank, a heat exchanger, two stages of pressure reduction, control, check and vent valves, and the necessary line as indicated in the schematic diagram Fig. 1-5. As can be seen from the general arrangement drawings Figs. 1-1 through 1-4, this system is quite compact and fits easily into the space between the tank assembly and the outer shell. A synthesis of the system is straightforward and requires principally selection of readily available components. Fabrication of the high pressure sphere is within the state of the art. The heat exchanger, however, will require a small amount of development but it is not a critical item.

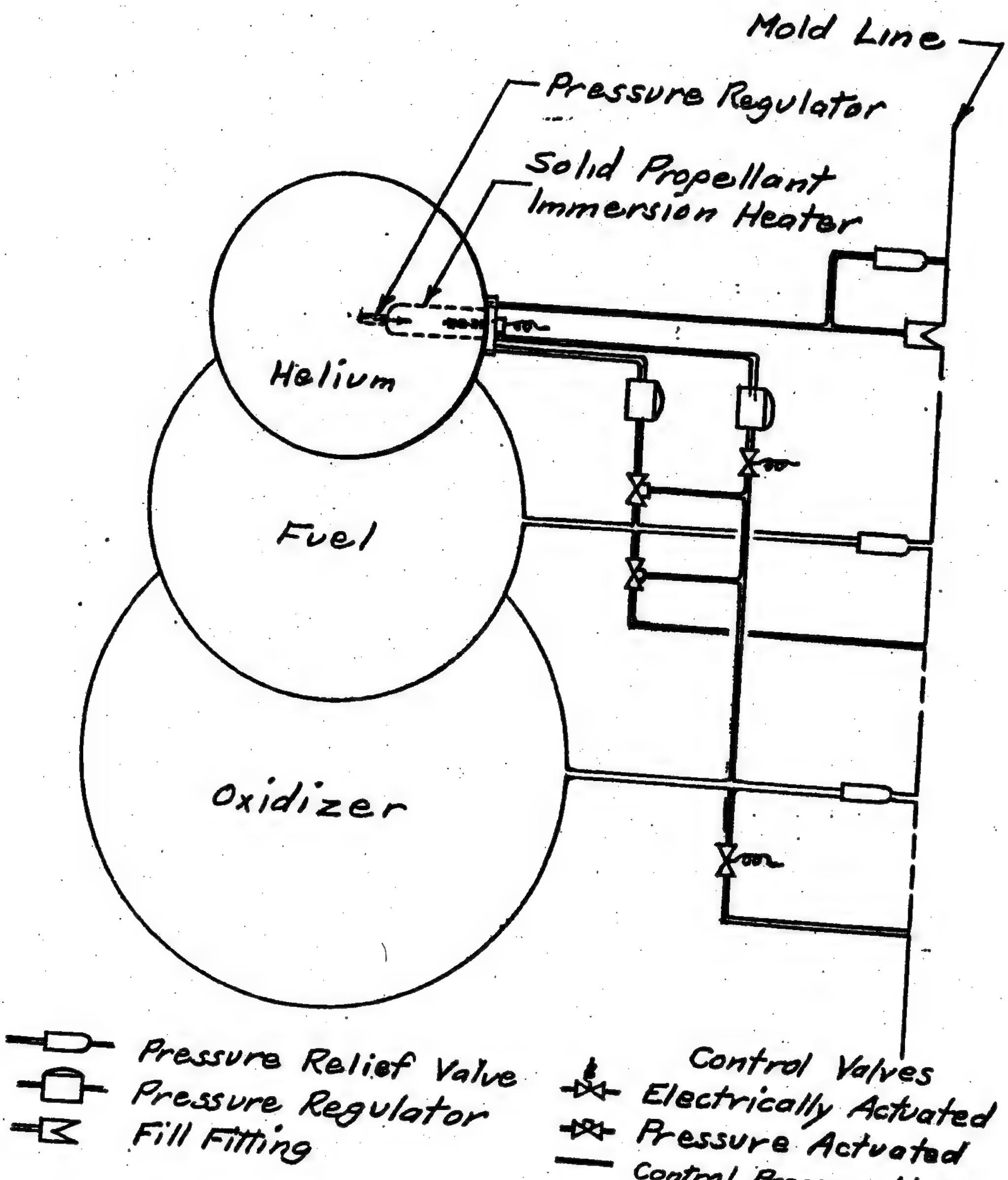

Fig.1-5 PIONEER PRESSURIZATION SYSTEM

Low level pressurization, required during fueling operations to insure integrity of the tanks, will be provided through the propellant overflow or vent lines. Pressure requirements for these preliminary functions will vary with the payload weights supported by the tanks and can readily be computed for each case. Elimination of the low pressure equipment from the airborne system not only reduces the weight but also enhances the reliability of the vehicle.

Proper sequencing of pressurizing (or venting) of the propellant tanks to maintain a higher pressure in the upper tank than in the lower one is necessary to prevent inversion of the intertank bulkhead. This is achieved by initiating the sequence with an electrically actuated valve which increases (or relieves) the pressure of one tank. When the pressure in that tank reaches a predetermined level, a pressure actuated valve automatically opens the inlet (or vent) line of the other tank.

The studies regarding environmental control for subsystems and equipment have not as yet been carried sufficiently far to permit showing of environmental control equipment or systems in the vehicle design. Environmental requirements have been established in orders of magnitude for some equipments. When a sufficient number of such requirements become known, a systems analysis can be made to determine the directions of design effort. It is known that the average temperature of equipments aboard the satellite will be of the order of 5°C. This indicates a need for generating and controlling additional heat. This may require combustion heaters and/or the use of solar energy. The need for using fluid and gaseous systems with their accompanying radiators, circulating pumps, pressurized reservoirs, and plumbing is anticipated. Extensive application of thermo-insulating and radiant heat reflecting or absorbing materials to the installations will undoubtedly be required. The question of multiple self-contained environmental control systems versus a central control system has not been resolved. Environment control by either compartments or single package has not been resolved in a firm manner. The vehicle design will lend itself without undue difficulty to most requirements. The vehicle skin can be made of steel, aluminum, or magnesium metal alloys, and radiators and skin can be integral (process of Western Roll-Bond Corp., East Alton, Ill.). If additional radiator surface is required, a portion of the adapter section of the vehicle-booster combination can be retained with the orbiting vehicle.

2. VEHICLE PERFORMANCE

The performance of the Atlas was investigated under the assumption that the present payload stage could be replaced with a final powered stage. The energy per unit mass of the final stage at sustainer burn-out was estimated for several weights of the final stage. The additional performance required of the final stage to put a payload in circular orbit at 300 nautical miles above the earth was determined as a function of initial energy per unit mass, gross weight, and burn-out weight, for a given propellant performance. In these calculations the final stage, for a given propulsion system, was assumed to coast to the point where firing would cause burn-out of the propellant at the instant orbital velocity and altitude were reached. The results of these calculations are shown in (Fig. 2-1). In this figure, energy E per unit mass of one body in orbit about its primary is given by

$$E = \frac{v^2}{2} - \frac{GM}{r}$$

where v is the velocity of the body relative to the center of mass of the two-body system, r is the distance of the body from this center of mass, and GM is a constant of the system involving the masses of the two bodies and the gravitational attraction constant. The energy E will have a negative sign for a body which does not escape from its primary. The gross weight is that of the stage going into orbit and the burn-out weight is the sum of the weights of payload, structure, and trapped propellant. The adapter and separation gear of this final stage was

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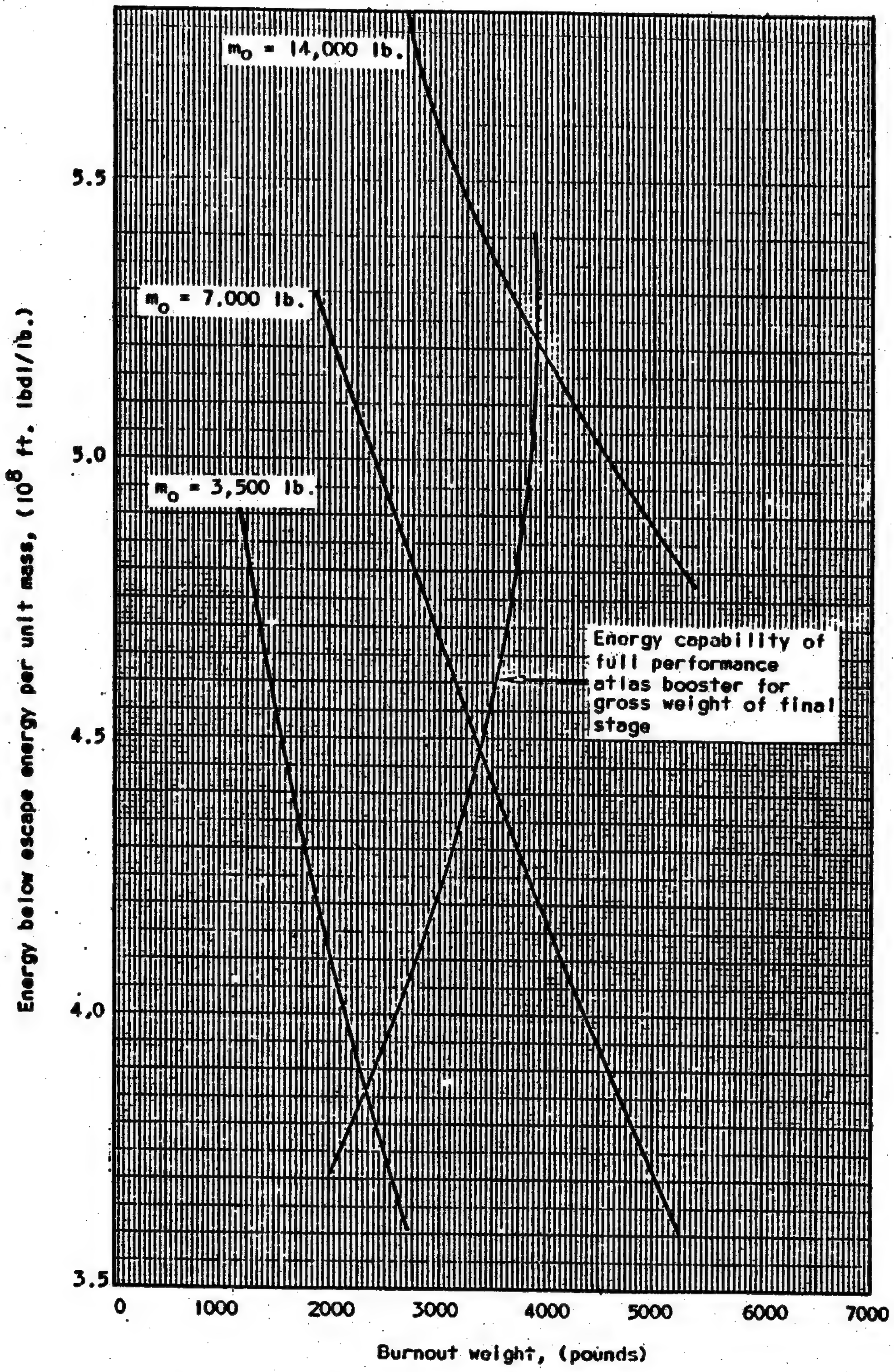

Fig. 2-1 Burnout Weight vs. Initial Energy per Unit Mass for Several Gross Weights (Isp = 300 sec.) and Capability of Atlas Booster.

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assumed to stay with the Atlas so that its payload weight was 160 pounds greater than the gross weight of the orbiting vehicle. From the impulse relationship

$$\Delta v = I_{sp} \; g \ln \left(1 - \frac{m_p}{m_o}\right)$$

where Δv is the velocity added by burning the propellant, I_{sp} is the specific impulse of the propellant, m_p is the mass of propellant, and m_o is the initial mass, then the change in mass of propellant Δm_p required for a change of specific impulse ΔI_{sp} is given by

$$\Delta m_p = -\Delta I_{sp} \; \frac{m_o - m_p}{I_{sp}} \; \ln \left(1 - \frac{m_p}{m_o}\right)$$

Using this relationship the burn-out weights for other propellant performances were determined.

Using the value shown in Ref. 1 the relationship of the energy per unit mass of the payload and the maximum range of the payload was determined. This was applied to the Atlas, where the undegraded Atlas was assumed to have a capability of 5,500 n. miles maximum range with a 3,500 lb warhead and the resulting total energies for several ranges were determined. Using these relationships and the burn-out weights of the Pioneer and Advanced vehicles, the weights of the payloads as functions of the initial gross weights and the Atlas range capability are shown in Figs. 2-2 and 2-3. Since both vehicles can accommodate any quantity of propellant less than the maximum, both can be used over a wide range of payloads for any performance of the Atlas. The preliminary design of the Advanced vehicle is discussed in Ref. 2 and illustrated in Fig. 3-6 in the same reference.

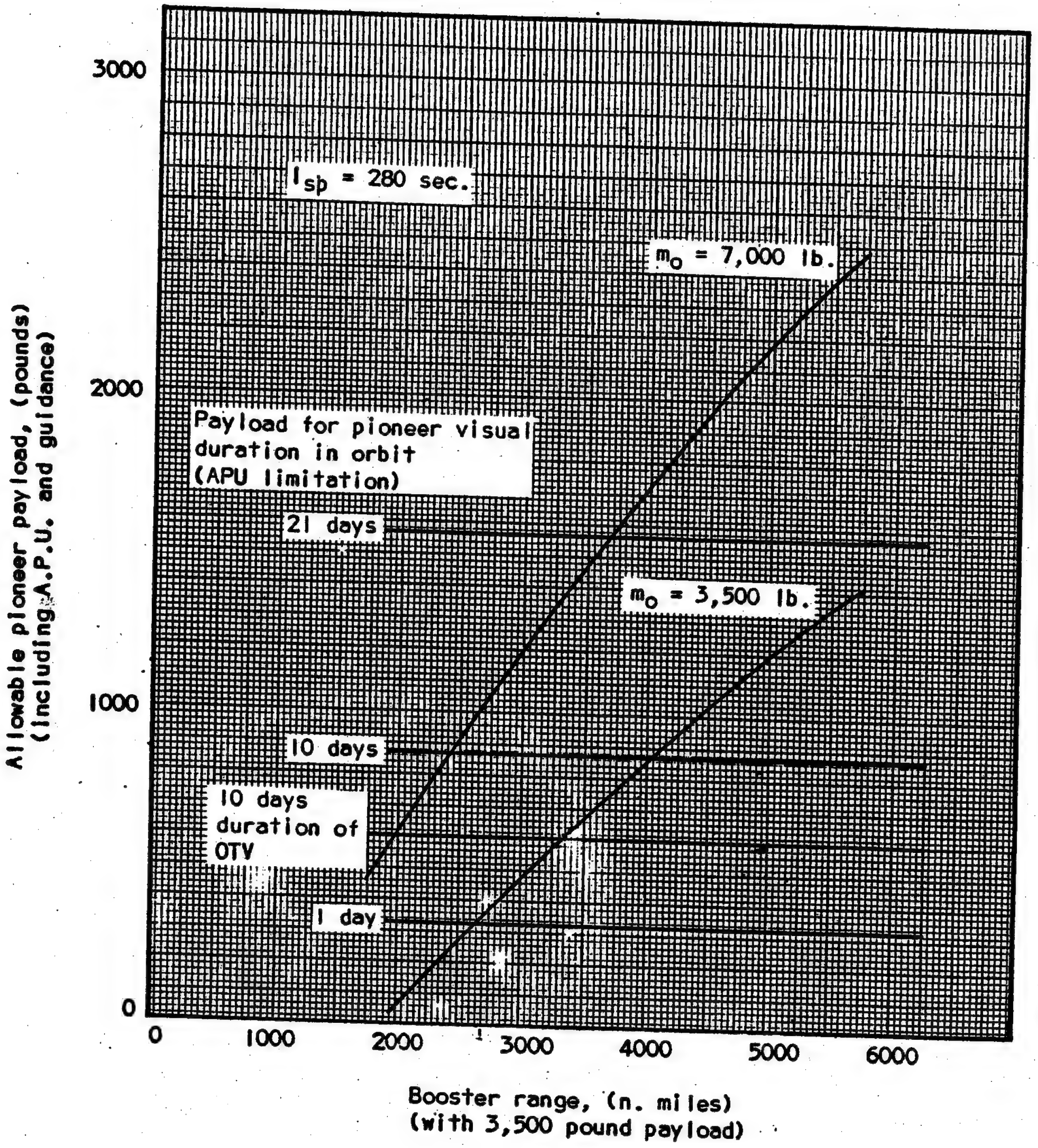

Fig. 2-2 Influence of Booster Performance on Payload and Duration of Pioneer

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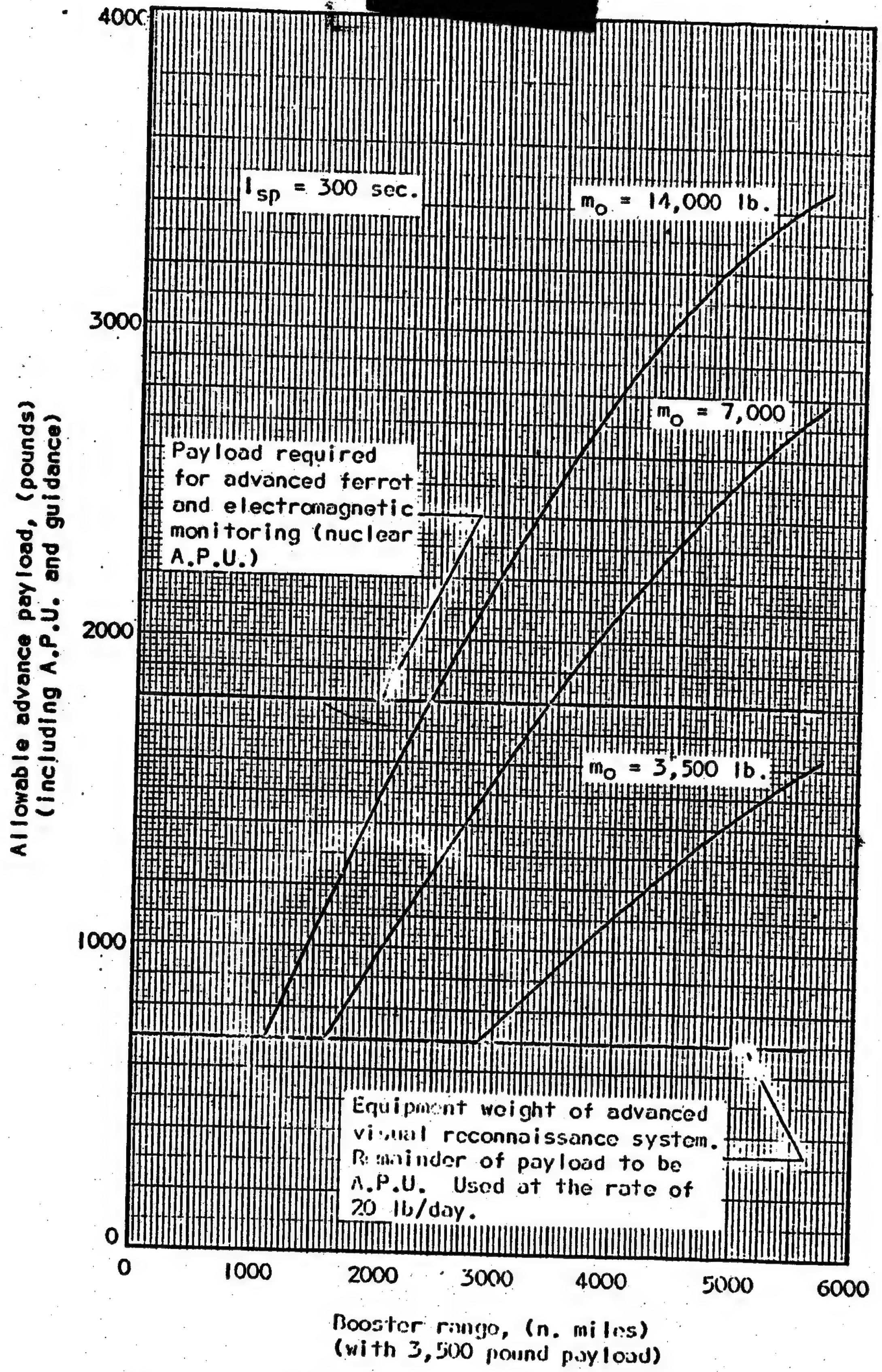

Fig. 2-3 Influence of Booster Performance on Payload and Payloads Required

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3. AERODYNAMICS

The aerodynamic studies for the PIED PIPER are essentially limited to the initial boost phase which is the only portion of the flight occurring in the atmosphere. Thus, the principal aerodynamic considerations are (1) the drag and stability of the ATLAS as affected by substitution of the OTV for the ATLAS warhead, and (2) the aerodynamic heating and airloads on the OTV as it is boosted through the atmosphere.

3.1 Drag

The drag of the PIED PIPER boost configuration was approximated by modifying an ATLAS drag coefficient curve to account for the difference in drag of the ATLAS warhead and a cone-cylinder OTV with a 14° half-angle cone. Using this derived drag coefficient curve and the trajectory data shown in Fig. 3-1, the drag impulse of the PIED PIPER was calculated and found to be approximately three percent of the thrust impulse. The drag of the OTV is only about eight percent of the total drag of the configuration; thus the drag impulse of the OTV is about 0.25% of the initial boost thrust impulse. Considering the small drag impulse of the OTV it is apparent that varying the configuration of the OTV within reasonable limits would not perceptibly affect the boost performance of the PIED PIPER.

3.2 Stability

The tip of the nose cone of the OTV extends nearly 11 feet farther forward than the warhead of the ATLAS which it replaces. This

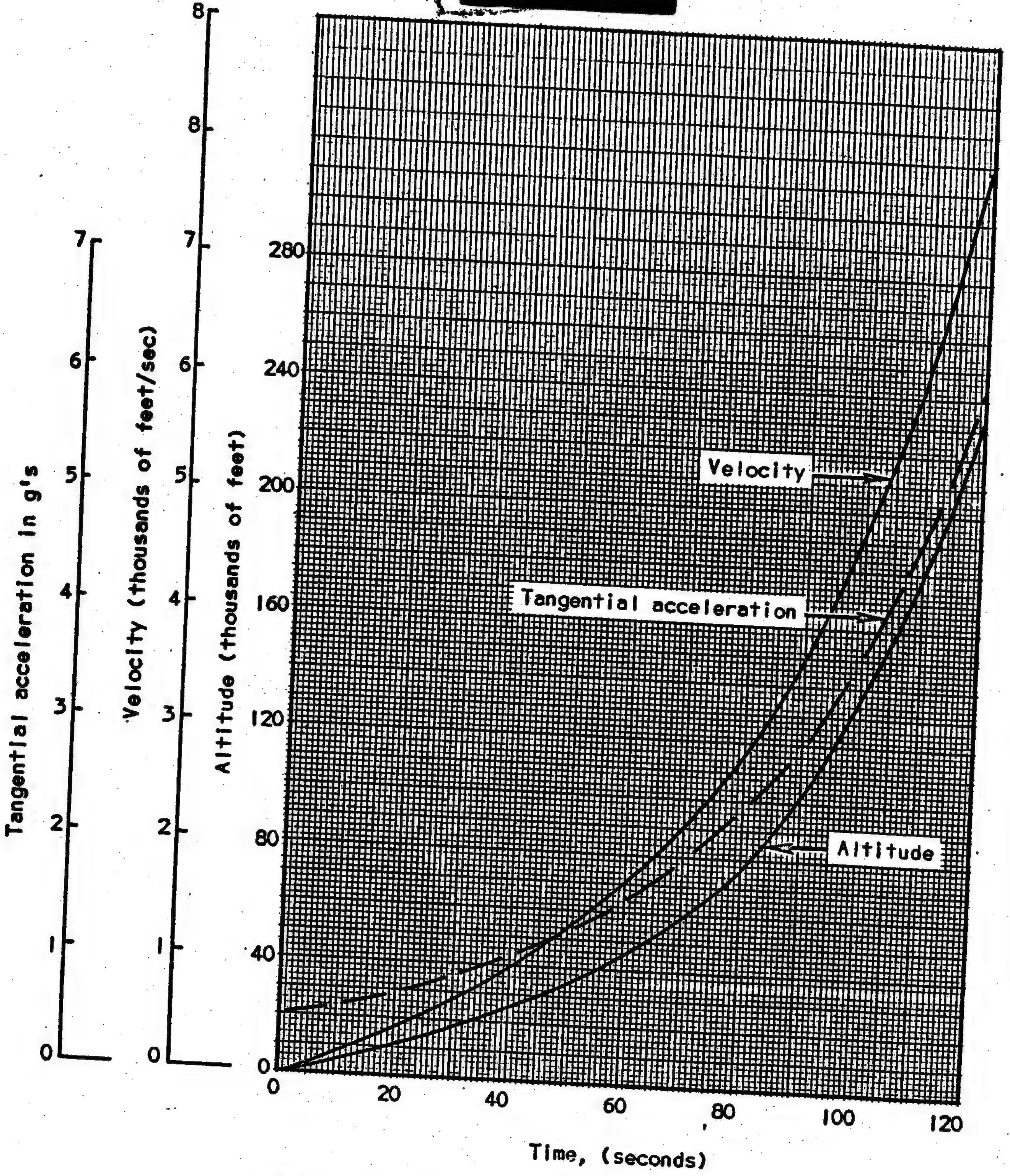

Fig: 3-1a Velocity, Altitude, and Tangential Acceleration Versus Time for O.T.V. Boost Trajectory

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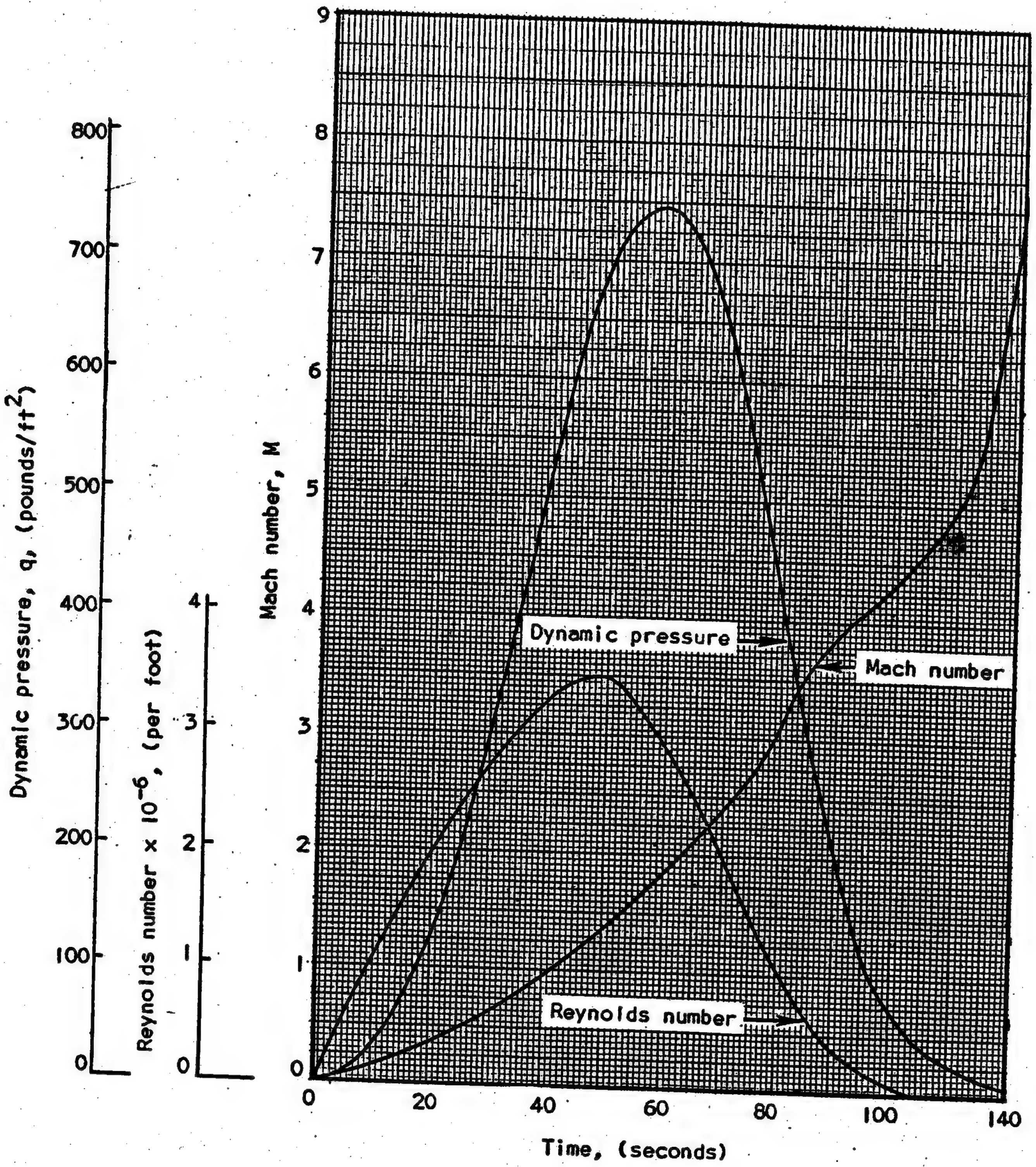

Fig. 3-1b Mach Number, Reynolds Number and Dynamic Pressure Versus Time for O.T.V. Boost Trajectory

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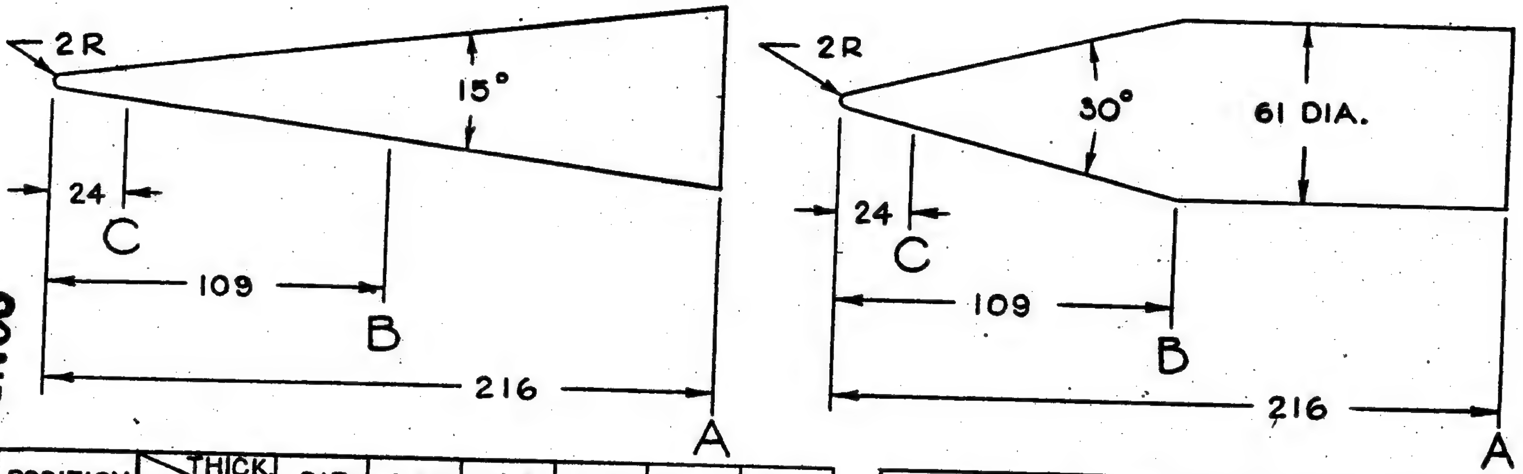
tends to move the center of pressure of the configuration forward but, because of the relatively small diameter of the OTV and the replacement of the spherically blunted warhead with a relatively sharp cone point, the displacement of the center of pressure is quite small. It is estimated that at Mach numbers greater than 1.0, the OTV moves the center of pressure of the ATLAS forward about 12 inches.

3.3 Air Loads

The load distribution on the OTV can be easily approximated because of the simple cone-cylinder configuration of the vehicle. Loading due to penetration of a sharp edged gust was estimated to provide data for calculating bending moments at various points along the OTV. A 150 foot per second gust at an altitude of 35,000 feet was assumed and the aerodynamic loads were calculated for the angle of attack of 6° caused by the gust encounter. This gust condition is rather extreme and is introduced to cause the maximum aerodynamic loads the structure will be expected to withstand.

3.4 Aerodynamic Heating

Skin heating analyses were performed at several different locations for two geometric configurations of the OTV. Sketches of the two configurations which are basically a 15° total angle conical body and a 30° total angle cone with a cylindrical afterbody are shown in Fig. 3-2. The locations for which the analyses were performed are indicated on these sketches.



POSITION	THICK. / MAT'L.	.015	.020	.035	.040	.050	.070
TEMP. @ A	AL. ALY.	☒	512	407	380	331	☒
	AM350	482	425	☒	☒	☒	☒
	MAG.	☒	☒	☒	☒	394	319
TEMP. @ B	AL. ALY.	☒	553	445	☒	☒	☒
	AM 350	522	465	☒	☒	☒	☒
	MAG.	☒	☒	☒	☒	☒	☒
TEMP. @ C	AL. ALY.	☒	638	531	500	451	☒
	AM350	607	550	☒	☒	☒	☒
	MAG.	☒	☒	☒	☒	514	439

NOTE: ALL TEMPERATURES IN °F.

.015	.020	.035	.040	.050	.070
☒	512	407	380	331	☒
482	425	☒	☒	☒	☒
☒	☒	☒	☒	394	319
☒	655	547	☒	☒	☒
624	567	☒	☒	☒	☒
☒	☒	☒	☒	☒	☒
☒	740	633	605	559	☒
709	652	☒	☒	☒	☒
☒	☒	☒	☒	618	549

Fig. 3-2 Calculated Maximum Skin Temperatures

Transient temperature calculations, using the trajectory shown in Fig. 3-1, were performed on an IBM 650 computer assuming various gages of stainless steel, aluminum alloy, and magnesium alloy for each location and configuration. A resume of the calculated peak temperature for each combination of parameters is presented in Fig. 3-2.

7 3 7 2 8 1

4. STRUCTURES

The structural loads, analysis, materials, dynamics and complete weight breakdown of the vehicle are treated in this section. Major structural design details are considered when they affect vehicle configuration and weight analysis.

4.1 Loads

4.1.1 Handling

The transportation equipment is designed to avoid stresses above those derived from flight conditions. Cradling and lifting are to be accomplished according to this principle.

Hoisting the vehicle onto the booster is accomplished by using an eyebolt and plate attached to the nose fairing structure. The ultimate hoist load factor is 3 at 15° pull-off.

4.1.2 Flight

The launching load factor of the complete system is 1.5. This is not a critical structural condition for this vehicle.

In designing the nose and skirt structure, a gust condition was investigated. Since the compressive strength of the outer shell is determined under other conditions, the vehicle is structurally capable of withstanding a 150 feet per second sharp edged gust ($\alpha = 6^\circ$). This would be very conservative.

The critical condition for the primary structure occurs at first stage burn-out. The tangential acceleration is at a maximum and is equal to (225 + 32.2) ft/sec^2 or 8 g's (see Trajectory of CMCC GM 281.1825; WD 56-00025). The side load factor was arbitrarily chosen to be 1.0 for design purposes; this combines a reasonable bending moment with the axial load. This condition is treated in detail in Sec. 5, Structural Analysis.

The separation of this vehicle from the booster presents no primary strength problems.

At the end of casting a design thrust of 10,000 pounds is experienced on this vehicle. This designs the engine mount in the compression sense and the tie in structure between the tanks and propulsion unit.

There are no structural problems at orbiting.

4.2 Structural Analysis

The primary structure consists of a stiffened shell, the spherical propellant tanks, and the propulsion unit mounting.

Three materials were considered for the shell structure: magnesium HK31A alloy, stainless steel AM 350, and an aluminum 6061ST. The materials must possess the following general properties: high buckling strength - weight ratio at 500°F., weldability, and panel flutter failure prevention. Magnesium HK31A excels on each of the above requirements. Mechanical properties are given in the sample calculation portion. There is no panel flutter with t = 0.070 with a panel length of approximately 22", since an internal shell pressure of 6 psi is contemplated, (Ref. 3).

If further study indicates that HK31A is not suitable from a temperature standpoint, an 0.020 AM 350 stainless steel under internal pressure can easily be utilized.

Stainless steel AM 350 is used for the high pressure spherical tanks. This steel exhibits almost 100% efficiency on welded joints; however, an 85% probability factor is used. At reasonable heat treats welded specimens were tested to 197,000 psi at room temperature. Corrosion resistance and high strength and creep rupture are of prime importance for this tank application.

Any high heat treat steel can be utilized for the propulsion unit mounting structure.

Four tank configurations, illustrated in Tables 6-15, 16, 17 of this Appendix, were considered utilizing a chamber pressure of approximately 310 psi. The configuration assumed a chemical APU tank mounted on top of the fuel tank. The weights of the tank proper structure including tank bottoms and rings are given in the weight analysis part. A typical calculation is given at the end of this section to demonstrate the superiority of spherical tanks over cylindrical tanks with conical tank ends. The hoop stresses of cylindrical tanks are twice those of spherical tanks. With well designed tank ends, the gage is three times that of a sphere. In addition, a hoop compression ring must be included to prevent tank collapse at the intersection of the tank end and side. A typical ring section is shown in the sample calculation portion of this section. The tank study shows that each cylindrical tank configuration weighs approximately 1,000 pounds against the 350 pounds for the spherical configuration.

The final tank configuration utilizing spherical tanks is shown in Paragraph 5.1. With the pressure-fed system, this is the most efficient structure that can be used. Mechanical properties, assumptions, and stress analysis of the primary structure are presented in the sample calculations.

4.3 Structural Dynamics

To establish the possibility of interaction of the elastic response of the vehicle and the dynamics of the control system, it is necessary to investigate the structural dynamics of the vehicle. In addition, this study is necessary to establish the dynamic loading of the structure of the orbiting stage and to investigate the possibility for the occurrence of dangerous structural resonances. For the sake of order and clarity, the study is divided into two parts according to the phase of the flight as follows:

1. Orbiting phase
2. Boost Phase

4.3.1 Orbiting Phase

In general, the orbiting phase can be divided into two sub-phases: (1) the separation problem which includes the events necessary to enter an orbit, and (2) the orbiting problem. It will be assumed that there are no excitations in the latter sub-phase; in the former there are many, the most important being due to:

1. The impulsive loading at separation
2. Motor starting
3. Maneuvering with the control motors

It is apparent that the main motor supporting structure is relatively flexible compared with the remainder of the orbiting stage structure. For purpose of analysis, the main motor assembly will be assumed to be one rigid mass connected by an elastic structure to another rigid mass comprising the tanks and equipment. While it is apparent that the motor assembly is relatively rigid, the assumption that the tanks are rigid requires some justification. It may be shown that complete and nearly complete thin, spherical shells vibrate only in modes which involve primarily stretching of the wall with consequent high energy storage, implying high natural frequencies. Thus, the tanks' structure, consisting of these types of shells, can have only relatively high natural frequencies suggesting that the rigidity assumption is valid.

Impulsive loading at separation will be primarily longitudinal, in which case there will be little expectation of difficulties. However, any lateral components may serve to excite lateral vibrations of the motor assembly relative to the primary structure of the orbiting stage. The mode will be considered in some detail below. The transient response of the structure due to motor starting has essentially the same nature as that due to separation impulse.

During maneuvering, two different excitations could be introduced by the control motors: pitch-yaw and roll corrections, Pitch-yaw corrections could excite a vibration consisting of a combination of lateral translation and rotation of the motor assembly relative to the primary structure. The lower natural frequency of this mode was found to be 6.4 cps. This calculation involved the use of Lagrange's equations wherein the potential energy included the strain energies due to bending, extension, and compression of the tubular motor supports. Including the requirements of conservation of linear and angular momenta reduced the problem to one of two degrees of freedom. In addition to pitch-yaw corrections this mode could be excited by lateral components due to separation impulse and motor starting.

Roll corrections can induce a mode wherein the motor assembly has a rotation about the vehicle axis relative to the primary structure. By a method similar to that used above the lower natural frequency in this mode was found to be 9.9 cps. In calculating this frequency it was assumed that the potential energy consisted only in that due to bending of the tubular supporting structure.

4.3.2 Boost Phase

It is anticipated that the control system of the orbiting stage may be required to monitor the flight during the boost phase.

Therefore, study of the structural dynamics of the entire vehicle (booster plus orbiting stage) during the boost phase is necessary to feed sufficient information to the control system of the orbiting stage to allow it to distinguish between elastic motion and vehicle maneuver. In addition, there exists the possibility of excessive structural loading within the orbiting stage as a result of the general vibrations of the vehicle.

Such problems can be solved, provided information concerning the modes of vibration of the entire vehicle is available. The fundamental and first harmonious modes were calculated with the results discussed below. The higher flexural modes may also be important, especially if vibration of components within orbiting stage are to be considered.

In order to arrive at the fundamental flexural mode shape and frequency, the vehicle was simulated by a free-free beam with mass and flexural rigidity distribution identical to that of the vehicle. Table 4-1 shows the distributions used in these computations. These data represent the best estimates available at the time of calculations. The actual calculations of the model shapes and frequencies were performed on the IBM 650 computer using a program based on Stodola's iterative method. The resulting model shapes and frequencies are plotted in Fig.4-1 for launch and booster burn-out configurations. It should be noted that the model shapes were normalized to the displacement of the forward end of the vehicle. Computer programs based on a modified Stodola routine are also available for computation of the higher modes.

Of the vibrational modes of the structure of the orbiting stage that could be excited by the general vibrations of the complete vehicle, two seem to be of primary importance: lateral and torsional vibrations of the motor assembly relative to the remainder of the vehicle.

In estimating the natural frequencies of these modes it was assumed that the motor assembly of the orbiting stage constituted a rigid mass connected by an elastic structure to an infinitely massive, rigid body which was assumed to simulate the remainder of the vehicle. The lateral mode is a combination of lateral translation and bending with a lower natural frequency of 5.2 cps. The torsional mode has a natural frequency of 8.6 cps.

The torsional mode could be excited by roll corrections or by coupling with torsional vibrations of the complete vehicle.

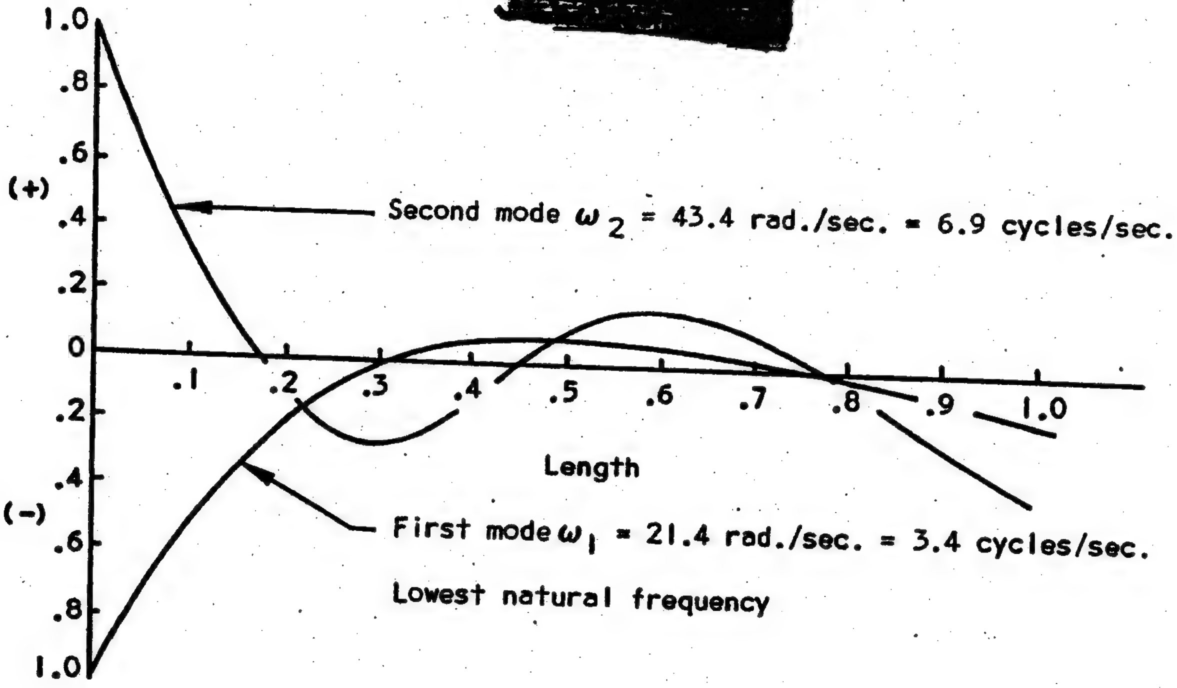
The lateral mode could be excited by pitch-yaw corrections of by coupling with lateral vibrations of the complete vehicle. From Fig. 4-1, it is seen that the vehicle has fundamental flexural normal modes of 3.4 cps at launch and of 9.9 cps at second-stage burn-out. Between launch and second-stage burn-out a mode will exist with a natural frequency coinciding with the figure of 5.2 cps calculated above for the lateral mode of the orbiting stage motor assembly. Obviously, the possibility of a dangerous resonance exists and further study is required to establish its severity.

TABLE 4-1

WEIGHT AND RIGIDITY DISTRIBUTIONS

(BOOSTER PLUS ORBITING STAGE)

ANALYSIS STA. NO.	FUS. STA.	EI(LB/N^2 x10^6) (LAUNCH)	(EI) BURNOUT 2ND STAGE	WT DATA w (LB/N) LAUNCH	WT DATA w (LB/N) BURNOUT 2ND STAGE
0	259	0	0	1.0	1.0
1	311	5000	5000	1.0	1.0
2	363	24780	24780	74.0	74.0
3	415	24780	24780	74.0	74.0
4	467	30000	30000	3.5	3.5
5	519	35500	35500	4.2	4.2
6	571	51900	51900	237.2	4.2
7	623	178000	178000	470.2	4.2
8	675	379980	379980	470.2	4.2
9	727	379980	379980	470.2	4.2
10	779	474960	474960	470.2	4.2
11	831	474960	474960	470.2	4.2
12	883	474960	474960	470.2	4.2
13	935	574960	574960	294.0	18.0
14	987	683960	683960	353.0	18.0
15	1039	683960	683960	353.0	18.0
16	1091	683960	683960	340.0	18.0
17	1143	7000000	30000	250.0	18.0
18	1195	7000000	30000	46.1	15.9
19	1247	7000000	30000	126.0	46.1
20	1299	7000000	30000	0.0	0.0



Complete booster burnout

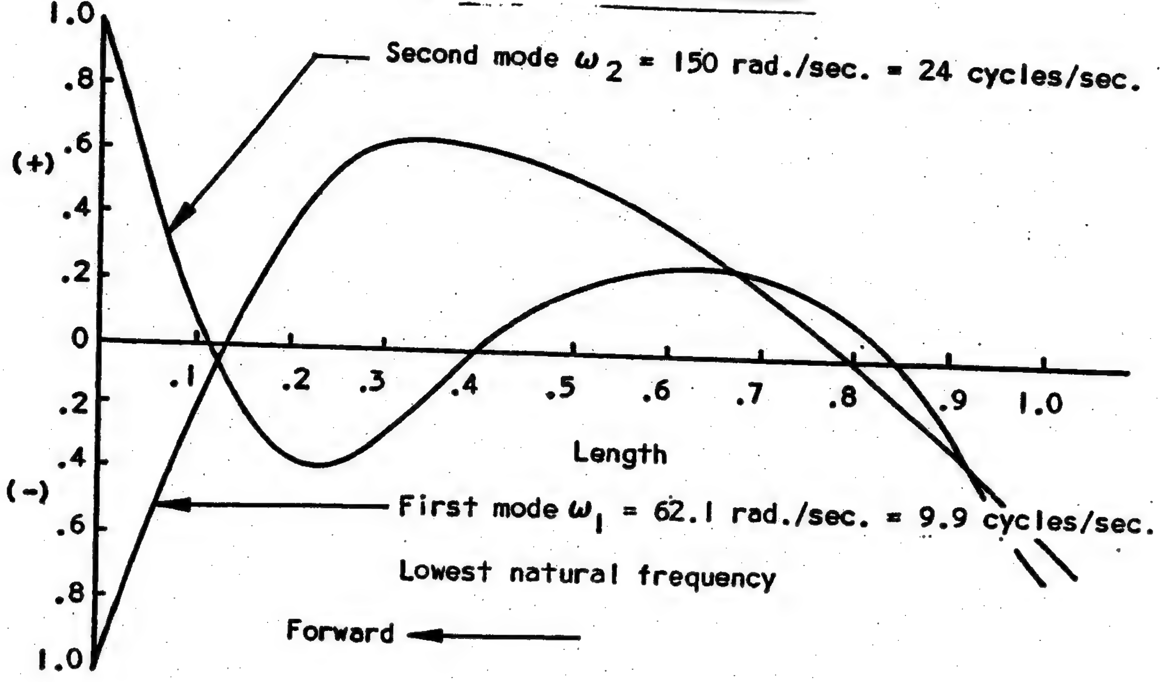
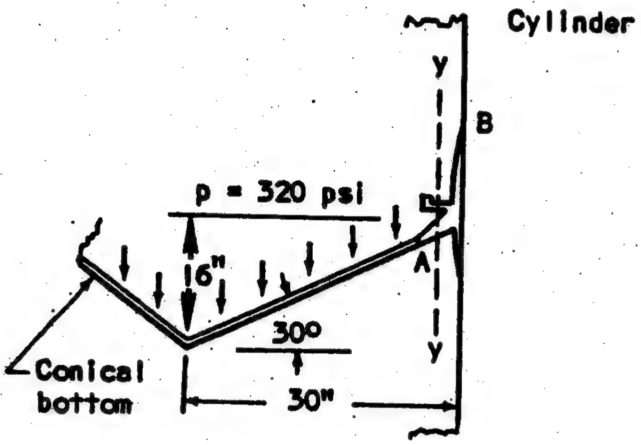


Fig. 4-1 Flexural Mode Shapes of Total Vehicle

4.3.3 Typical Conical Tank Bottom Structure



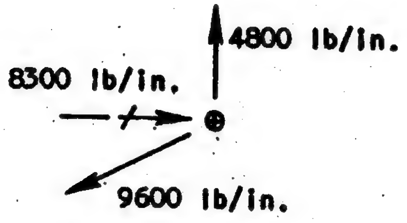
sin 30° = .5; tan 60° = 1.73

Circumference = 189 in.

Cross sectional area = 2830 in^2

Vertical load = 2830 x 320
= 910,000 lb

Loading = $\frac{910,000}{189}$ = 4800 lb/in



Along cone = $\frac{4800}{\sin 30°}$ = 9600 lb/in longitudinally.

This induces compression in the ring due to change in direction.

Hoop tension relief:

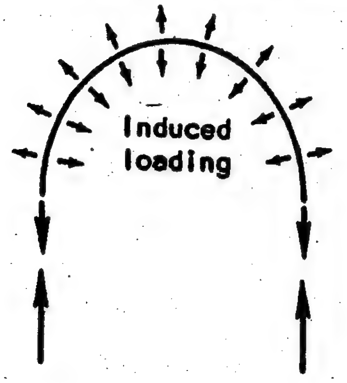
Conical at A = $\frac{320 \times 16 \times 1.73}{15}$ = 17,700 lb/in

Cylindrical at B = $\frac{320 \times 60}{2}$ = 9600 lb/in

Assume average x length A to B is relief load in ring.

Average is 13650 lb/in and ℓ = 3.6 in.; therefore, load is 47,800 lb.

Hoop tension relief loading



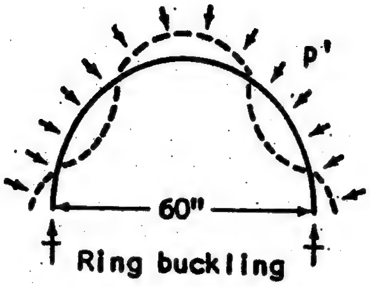
47,800 lb

$$P = \frac{8300 \times 60}{2} = 248{,}000 \text{ lb}$$

Net load in ring is 248,000 - 47,800 = 200,000 lb

$$\text{Net } p' = \frac{200{,}000}{30} = 6700 \text{ lb/in.}$$

$p' = \frac{3EI}{r^3}$ critical loading for ring buckling

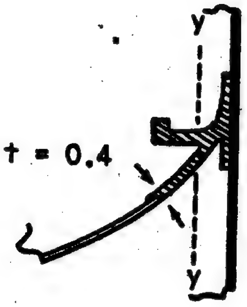


$$I_{\text{required}} = \frac{6700 \times 27000}{3 \times 30 \times 10^6}$$

y - y

$$= 2.02 \text{ in}^4$$

for this particular pressure and angle.



4.4 Structural Testing

The structural testing of this vehicle is to be performed as described below. In general, Lockheed conducts all tests except the proof loading of the spherical tanks; this is to be done by the tank manufacturer. Any testing with tanks pressurized must be performed under explosive conditions, since the pressures are so great.

The miscellaneous tests consist of engine mount, hoist, battery case, welded magnesium and AM 350 steel specimens, natural frequency of engine and tank combination, and any other necessary structure. Any heat simulation can be attained by means of radiant energy lamps.

The gust condition (altitude $h = 35{,}000$, $U = 100$ ft/sec) is critical for the nose shell and nose-tank attack structure. This test is to be simulated by means of a whiffle tree attached to the shell with tanks pressurized. The latter is necessary since the bending must be taken by the tanks between the cone base and the separation plane as noted in the sketch in paragraph 5.3.

The maximum longitudinal load factor condition at first stage burn-out is tested last. This condition is critical for most of the structure and is the failing load test. The nose cone may be removed during this test in order that simulated load can be imposed on the equipment racks. All load is generated by means of hydraulic jacks and tension straps. The skirt of the vehicle is so mounted as to simulate booster attach.

5. STRUCTURAL ANALYSIS

The following assumptions have been made:

1. Stress formulae from Ref. 6.
2. Load factor of 1.5 for any ground type conditions and tank design since they are pressurized on the ground. The load factor of flight conditions is 1.25.
3. Weld efficiency is 85%.
4. U.T.S. for AM350 stainless thin sheet at room temperature is 198500 psi. Limit allowable stress is therefore:

$$\frac{198500 \times .85}{1.5} = 112{,}200 \text{ psi} \qquad \text{(Ref. 4)}$$

5. For thick members, such as rings, U.T.S. of AM350 is 160,000 psi.
6. Mechanical properties of HK31A magnesium (Ref. 5) are in Figs. 5-1 and 5-2.
7. Compression on curved sheet is conservatively estimated as

$$9E\left(\frac{t}{b}\right)^{1.6} \qquad \text{(Ref. 6)}$$

8. Design gross weight W_G of the orbiting vehicle is 3500 lb for all structure except the spherical tanks and hoist fitting.

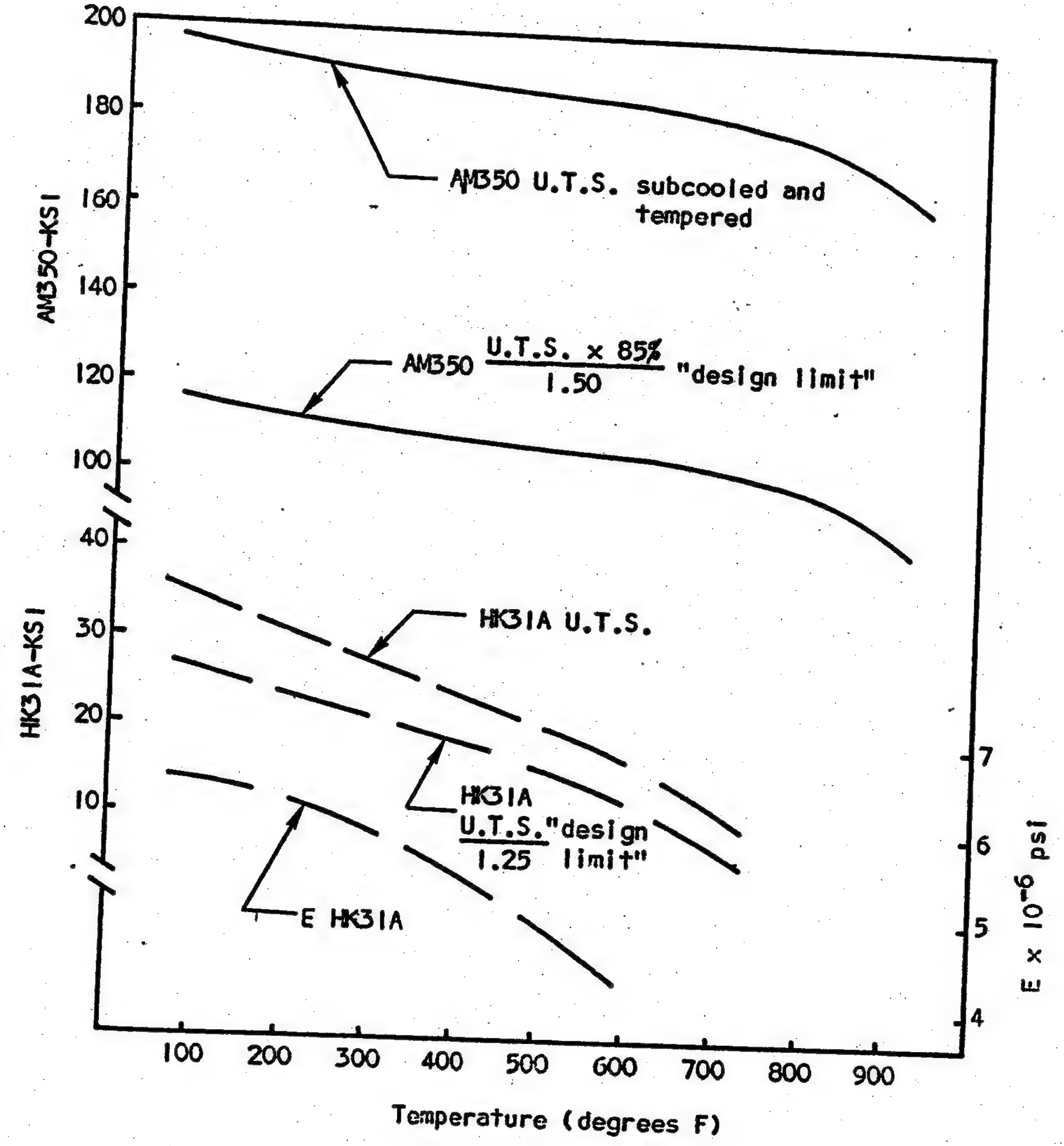


Fig 5-1. Mechanical Properties of AM350 Stainless Steel and HK 31A Magnesium Alloy

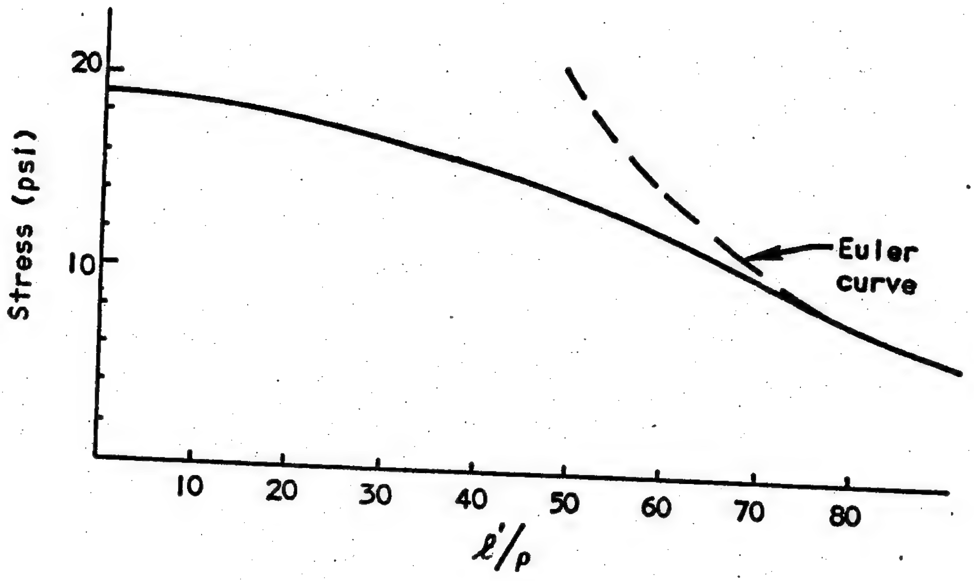
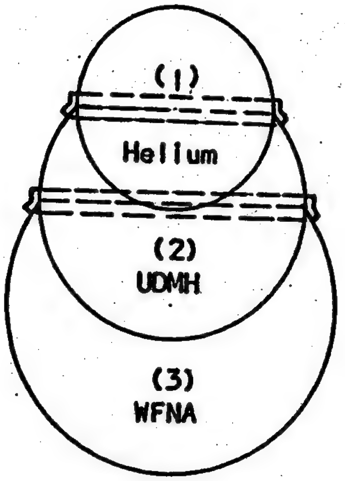


Fig 5-2. Column Allowable for HK 31A Magnesium at 430°F

5.1 Tank Structure (Final Configuration)

CONDITION:



TANK DATA:

(1) $p = 3000$ psi
$d = 32"$
$V = 10.2\ \text{ft}^3$

(2) $p = 315$ psi
$d = 44.2"$
$V = 19.6\ \text{ft}^3$

(3) $p = 300$ psi
$d = 54"$
$V = 29\ \text{ft}^3$

Required gages for strength

utilizing AM350 stainless steel:

$$t = \frac{p\,d}{4 \times 112{,}200}$$

(1) $t = \dfrac{3000 \times 32}{4 \times 112{,}200} = .213$ USE .250

(2) $t = \dfrac{330 \times 44.2}{4 \times 112{,}200} = .0324$ .036

(3) $t = \dfrac{329 \times 54}{4 \times 112{,}200} = .0396$ .045

NOTES: 1. Pressures include head and assumed tolerance of 5 psi

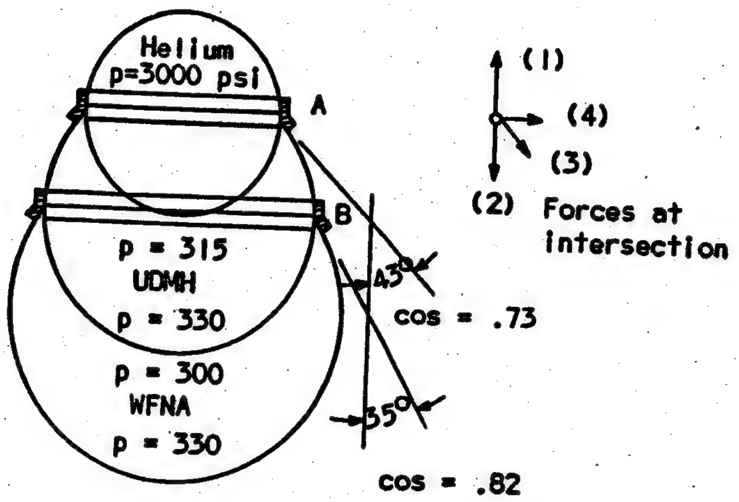
2. Gage on tank (3) increased because of engine mount attach.

5.2 Tank Intersection Rings

Condition: Burnout of first stage, $W_G = 6500$ lb,

$h = 200,000$, $n_z = 8.0$, $n_x = 1.0$, Tank Design

Pressures as Shown.



Helium Tank: d = 32 in

Cross Sectional Area = 804 in^2

Circumference = 101 in

UDMH Tank: d = 44.2 in

Cross Sectional Area = 1530 in^2

Circumference = 139 in

Tension Ring at A

Loading (1) $\frac{p \times \text{Area}}{\text{Circumference}} = \frac{3000 \times 804}{101}$

Loading (2) $= 23{,}800$ lb/in

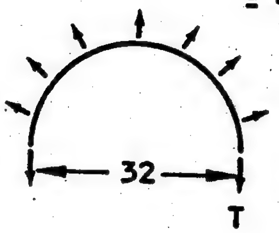
$$\frac{p \times \text{Area}}{\text{Circumference}} = \frac{2685 \times 804}{101}$$

$$= 21{,}300 \text{ lb/in}$$

Loading (3) $\frac{315 \times 804}{101 \times .73} = 3450$ lb/in

Loading (4) $3450 \sin 43^\circ$

$= 2350$ lb/in



$$T = \frac{2350 \times 32}{2}$$

$$= 37{,}600 \text{ lb}$$

Area required $= \frac{37600 \times 1.5}{160{,}000} = .352 \text{ in}^2$ at A

Tension Ring at B

Loading (1) $\frac{330 \times 1530}{139} = 3630$ lb/in

Loading (2) $\frac{30 \times 1530}{139} = 330$ lb/in

Loading (3) $\frac{300 \times 1530}{139 \times .82} = 4030$ lb/in

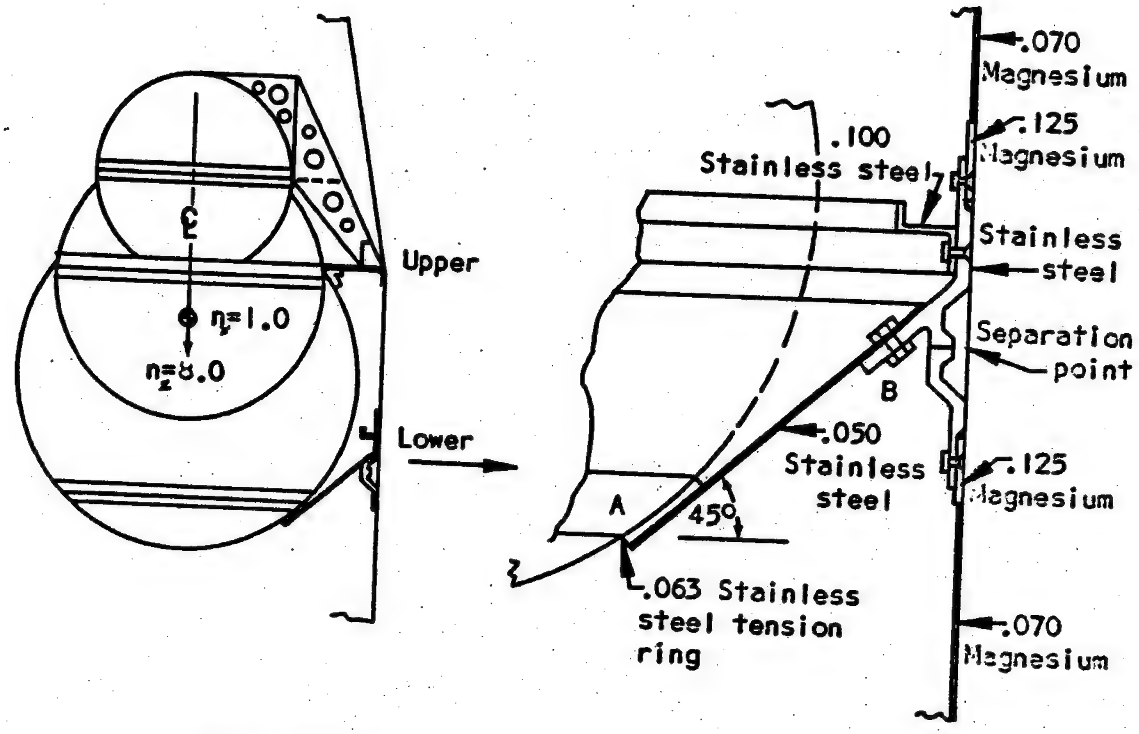
$$\frac{4030 \times 44.2}{2} = 89{,}000 \text{ lb}$$

Area required $= \frac{89{,}000 \times 1.5}{160{,}000} = .836 \text{ in}^2$ at B

5.3 Tank Attachments to Outer Shell

Condition: Burn-out of first stage, $W_{cg} = 6500$ lb

$h - 200,000$, $n_z = 8.0$, $n_x = 1.0$



Lower Attach

Assume lower attach carries entire vertical load:

$W_I = W_G$ -(Weight of shell)

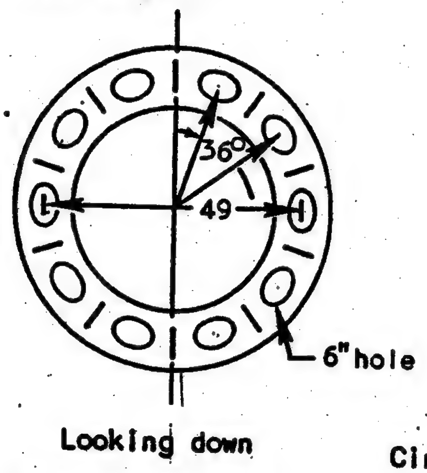
$W_I = 6500 - 300 = 6200$ lb, Internal load

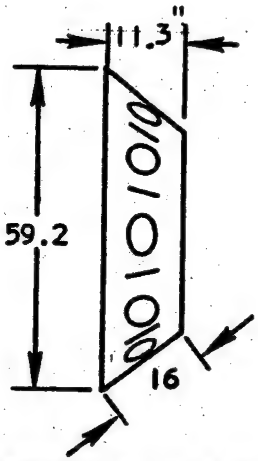
Vertical loading at A $= \frac{6200 \times 8}{60} = 825$ lb/in

Design loading along A-B is $= \frac{825 \times 1.25}{.707} = 1460$ lb/in

[redacted]

Cone Adaptor

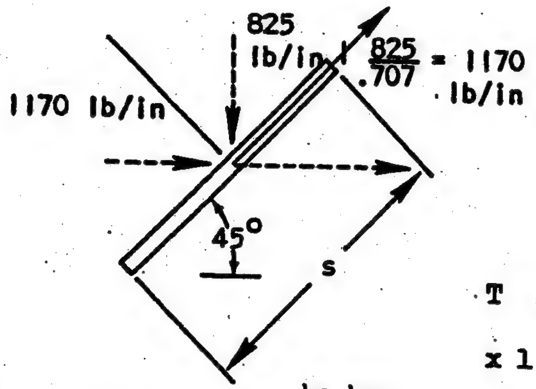


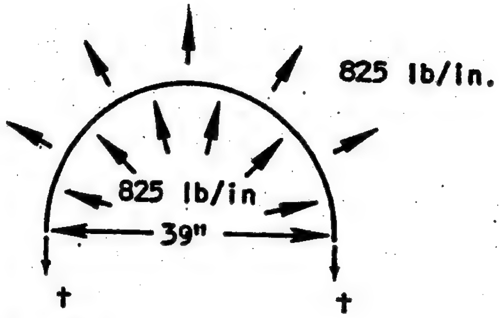


Circumference = $\pi(49) = 154"$

This adaptor also carries some side load.

Tension Ring (assuming no weld to tank)





$$T = \frac{825 \times 2 \times 39}{2} = 32{,}200 \text{ lb}$$

$$\times 1.25 = 40{,}400 \text{ lb}$$

$$\text{Required Area} = \frac{40{,}400}{160000} = .252 \text{ in}^2$$

Internal pressure is 325 psi minimum for this condition.

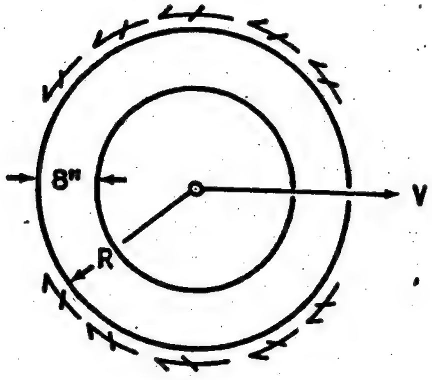
Circumferential loading is $1170 \times 1.25 = 1460$ lb/in.

Distance S must be at least $\frac{1460}{325} = 4.5$ in. Ring is therefore $4.5 \times .063 = .283 \text{ in}^2$.

[redacted]

Upper Tank Attach

Assume upper attach carries entire side load:



Upper attach ring

$V = 6200 \text{ lb} \quad R = 30"$

$$q_{max} = \frac{6200}{\pi R} \quad \text{shear flow}$$

$$= 65.4 \text{ lb/in.}$$

$$F = K \frac{\pi^2 E}{1^2(1-\mu^2)} \left(\frac{t}{b}\right)^2 \quad \text{Shear buckling}$$

With $t = .040$ $F_{buckling} = 7.4 \times 30 \times 25 = 5550$ psi

$$\tau = \frac{65.4}{.040} \times 1.25 = 2050 \text{ psi}$$

A minimum of .040 is required for rigidity.

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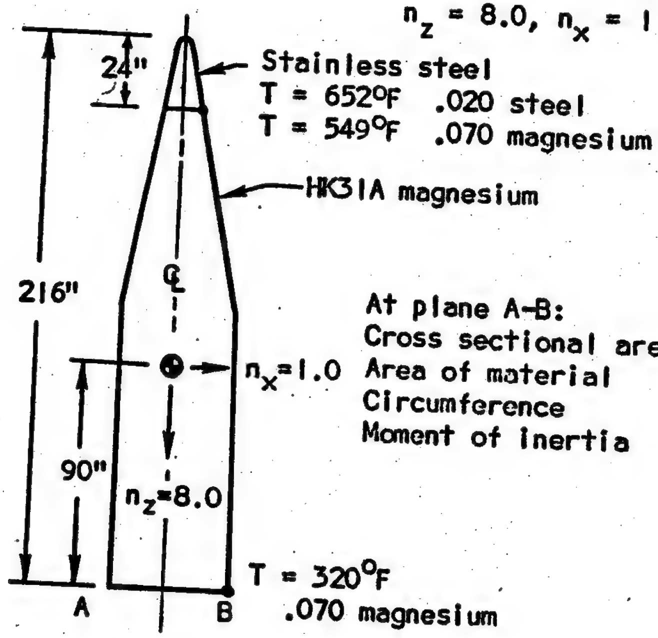
[redacted]

5.4 Shell Structure

Condition: Burnout of first stage, W = 6500 lb, h = 200,000', $n_z = 8.0$, $n_x = 1.0$



At plane A-B:
Cross sectional area = 2830 in^2
Area of material = 13.4 in^2
Circumference = 189 in
Moment of inertia = 5900 in^4

With internal pressure of 6 psi:
Axial load = 3500 x 8 = 28,000 lb

Bending moment = 3500 x 90 = 316,000 lb

$$\text{Relieving load} = \frac{\pi \overline{60}^2}{4}(6) = 17{,}000 \text{ lb}$$

$$\sigma_{A-B} = \frac{(28000 - 17000)}{13.4} + \frac{316000 \times 30}{5900} = 2430 \text{ psi}$$

t = .070

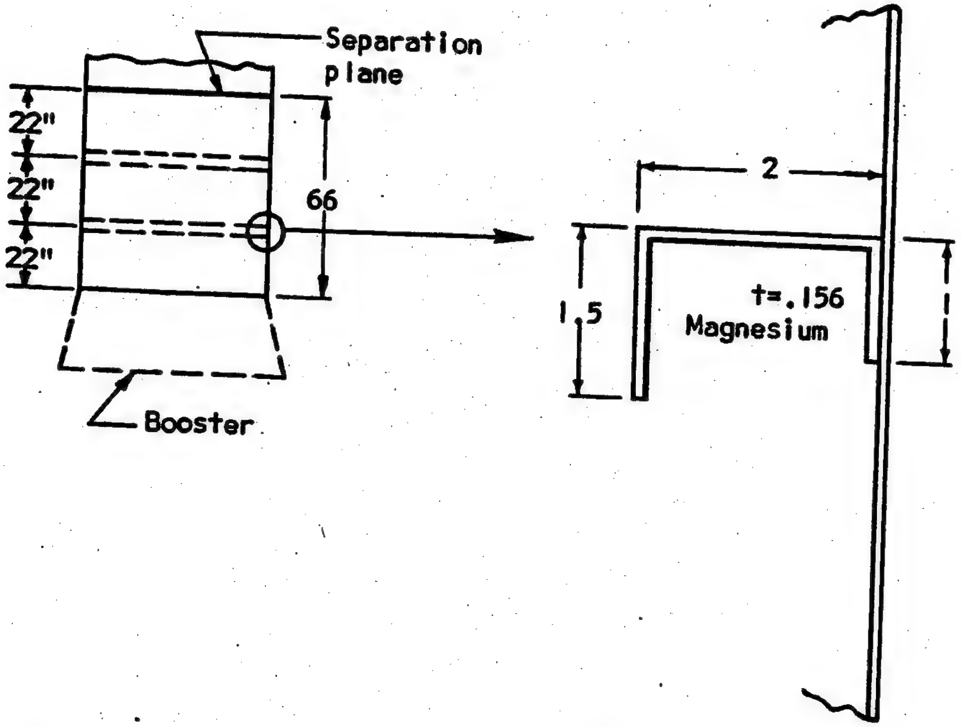
$$\times 1.25 = 3040 \text{ psi}$$

$$F_{buckling} = 9 \times 5.6 \times 10^6 \times \left(\frac{.070}{30}\right)^{1.6} = 3280 \text{ psi}$$

~~CONFIDENTIAL~~

[redacted]

Lower Portion of Shell



These frames act as stabilizing members against buckling of the skin

$$\frac{\text{Area of stiffner}}{\text{Area of web}} = .33$$

Lower Portion of Shell without Internal Pressure

t = .080 magnesium for panel flutter requirements

$$F_{buckling} = 9 \times 5.4 \times 10^{6} \left(\frac{.080}{30}\right)^{1.6} = 3640 \text{ psi}$$

$$A = \pi dt = \pi 61 \times .080 = 15.4 \text{ in}^2$$

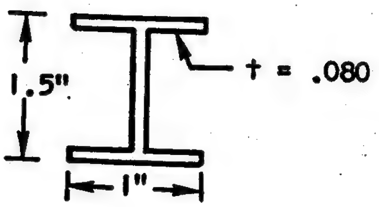
$$I = \pi r^3 t = \pi 30^{-3} \times .080 = 6800 \text{ in}^4$$

Since axial and moment loads act at the same time and t = .080 is a requirement.

$$1.25 \left[\frac{28000}{15.4} + \frac{316000 \times 30}{6800}\right] x = 3640 \text{ psi}$$

x = 90% ultimate taken by .080 cylinder
10% by stiffness

1.25 (28000) x 10% = 3500 lb
1.25 (316000) x 10% = 39,600 lb



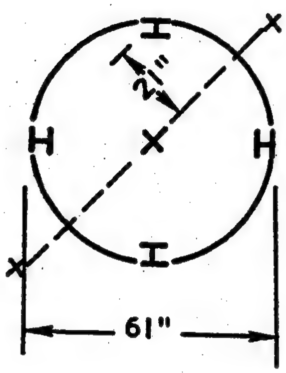
$$A = .280 \text{ in}^2 \quad \frac{\ell}{\rho} = \frac{22}{.615} = 35.8$$

$= .615$ in $\quad F_c = 15,000$ psi (See paragraph 5.1)

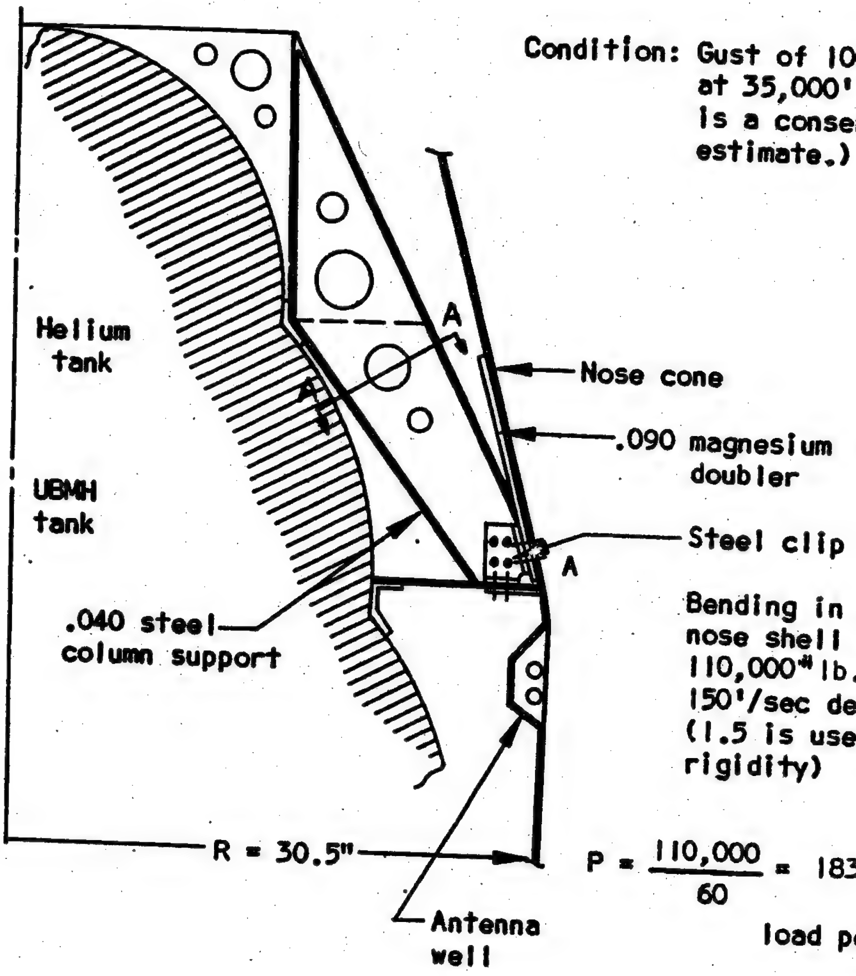
I_{xx} is minimum at 492 in^4

$$\sigma = \left[\frac{3500}{4(.28)} + \frac{39600 \times 30}{492}\right] 1.25 = 6920 \text{ psi}$$

These stiffners would therefore be adequate.



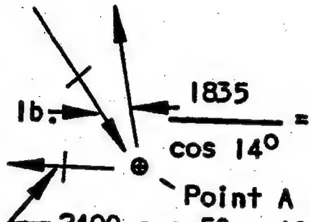
5.5 Nose Cone Support Structure



Condition: Gust of 100'/sec at 35,000' (This is a conservative estimate.)

Bending in the nose shell equals 110,000"lb. from 150'/sec design (1.5 is used for rigidity)

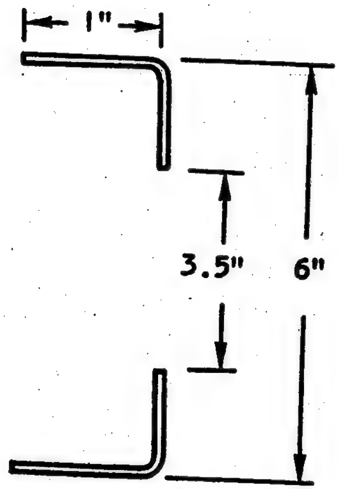
$P = \frac{110,000}{60} = 1835$ lb. Total load per side



$\frac{1835}{\sin 50^{\circ}} = 2400$ lb.

$\frac{1835}{\cos 14^{\circ}} = 1900$ lb. from shell

$2400 \cos 50 - 1900 \sin 14^{\circ}$
$= 1080$ lb. into the ring



Section A-A

ℓ = 18 in

t = .040

A = 4.5 x .040 .180

$\sigma = \frac{2400}{.180} = 13{,}300$ psi

$I_{yy} = .011 \text{ in}^4$

$\rho = \sqrt{\frac{.011}{.180}} = \sqrt{.061}$

$= .247$

$\frac{\ell}{\rho} = \frac{18}{.247} = 73$

$\frac{b}{t} = \frac{1}{.040} = 25$

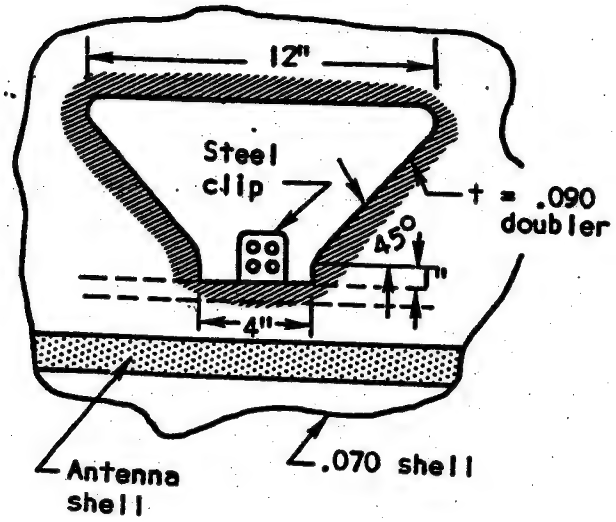
Johnson - Euler

$F_{buckling}$ = 20,000 psi

F_{column} = 18,000 psi

c = 1.5

Magnesium doubler



A = 12 x .070 = .84

$\sigma = \frac{1835}{.84} = 2200$ psi

$F_{c_{.070}}$ = 2800 psi

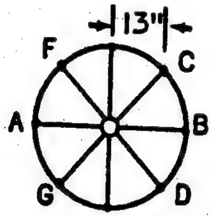
Assume doubler takes all the load at lower end.

$\sigma = \frac{1835}{4 \times .090} = 5100$ psi

$F_{c_{.090}} = 9 \times 4.8 \times 10^6 \left(\frac{.090}{30}\right)^{1.6}$

$= 5200$ psi

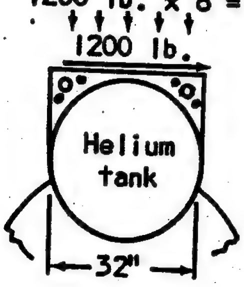
5.6 Equipment Support Structure



Condition: Burn-out of first stage, $W_G = 3500$ lb
$h = 200{,}000$ ft
$n_z = 8.0$, $n_x = 1.0$

$M = 1200 \times 22 = 26{,}400$" lb.

1200 lb. × 8 = 9600 lb.



Loads into the helium tank are stabilized laterally by the internal high pressure.

Compressive loading on brackets is critical:

Assume AB takes all shear:

$$q = \frac{1200}{32} = 37.5 \text{ lb/in.}$$

Vertical loading on each support:

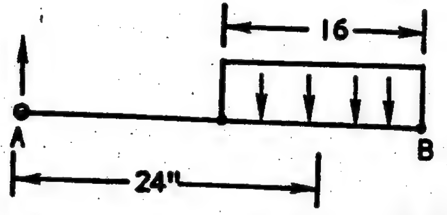
$$L_v = \frac{9600}{8 \times 16} = 75 \text{ lb/in}$$

Assume the bending is taken by uniform loading along OB, OC, and OD (since flexibility increases going from center to edge of platform),

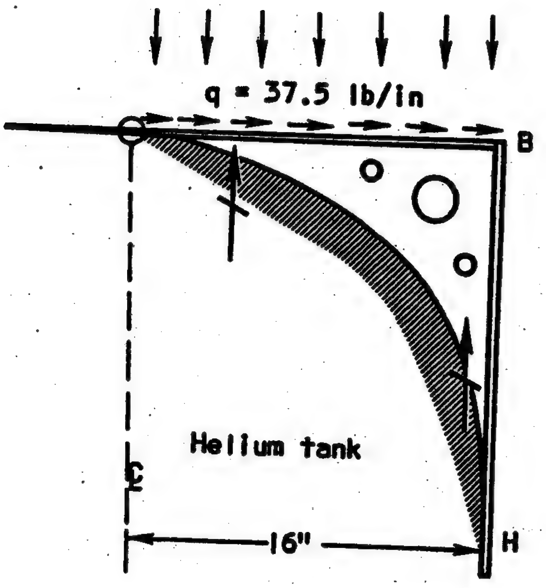
$$OB = \frac{26400 \times 16}{42} = 10{,}000 \text{ lb}$$

$$OC = OD = \frac{26400 \times 13}{42} = 8{,}200 \text{ lb}$$

The tension loading due to moment is taken by the lash down straps at opposite points, A, F, and G



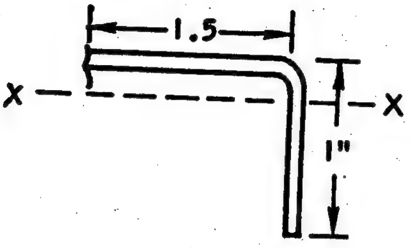
Loading due to moment $= \dfrac{10,000}{24 \times 16} = 26$ lb/in



Design ultimate vertical loading = 1.25 (26 + 75) = 126 lb/in

Total load = 126 × 16 = 2010 lb.

Assume OB is a beam supported on each end (conservative), Column BH must take 2010/2 = 1005 lb



$t = .040$ $A = .100$ $I_{x-x} = .006$ restrained from twisting

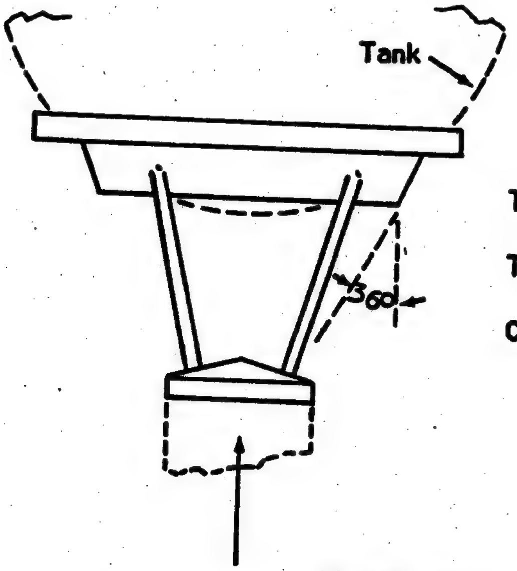
$\ell = 15$ $\rho = .245$

$F_{buckling} = 20,000$ psi $\dfrac{\ell}{\rho} = 61$

$\sigma = \dfrac{1005}{.100} = 10,000$ psi $F_{column} = 18,000$ psi

Johnson-Euler

5.7 Engine Mount Structure



True length, 10 inches

True angle, 36°

Cos 36° = .81

Each member takes $\frac{10,000}{4 \times .81} = 3100$ lb. × 1.25 = 3880 lb.

Requires t = .036 1" O.D. 4130 (HT 95,000 psi) steel tubing.

6. WEIGHT AND BALANCE

This section includes detailed information for the Pioneer Orbiting Test Vehicle, Pioneering Visual Reconnaissance Vehicle, and the Pioneer Ferret Reconnaissance Vehicle, and contains the following weight comparison studies:

1. Conical vs. ogive nose section
2. Pressurized vs. non-pressurized aft section
3. Discussion - Pressurization system weight comparison
4. Propellant tank studies (high vs. low pressure)
    a. UDMH & WFNA
    b. JP-4 & LOX

Table 6-1 provides a summary of weights for the three vehicles. Moments of inertia for the Pioneer Ferret Reconnaissance Vehicle are listed in Table 6-2.

Payload weights for the three vehicles are given in Tables 6-3 through 6-5.

In the weight empty breakdown in Tables 6-6 through 6-11, the propellant tanks have not been listed under structure for the convenience of weight comparison. However, as all tanks are primary load-carrying encumbers, they should be considered as structures.

Table 6-12 presents a weight comparison of the nose section for conical and ogive configurations. Table 6-13 presents a weight comparison for the skirt section (pressurized versus non-pressurized). In these comparisons identical values are omitted; only divergent values are recorded.

Table 6-14 presents a comparative weight study of the pressure-fed propulsion system and the pump-fed propulsion system. Both systems are based on 3,560 lb total propellant.

Tables 6-15 through 6-17 present a comparative weight study of four propellant tank configurations for a system using LOX as oxidizer and JP-4 as fuel. This study was based on 4,340 lb total propellant.

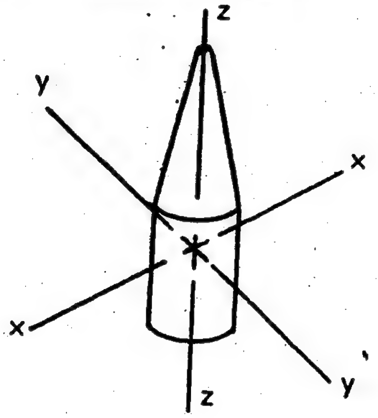
6.1 Weight

The weight figures for the structure and propellant tanks assembly are based entirely on the stress analysis included in the structures section of this appendix.

6.2 Moments of Inertia

The reference axis for moments of inertia are shown in the sketch. Roll moments of inertia are about the Z-Z axis. Pitch and yaw moments of inertia were assumed to be identical.

The Pioneer Ferret Reconnaissance Vehicle was used as a typical example for the moment of inertia calculations.



6.3 Center of Gravity

The longitudinal reference datum is located at the extreme tip of the nose hemisphere, which is fuselage station zero. The extreme aft end of the vehicle is fuselage station 216.0. These stations are common to all pioneer vehicles. A maximum center of gravity travel occurs on the Pioneer OTV version. The center of gravity moves from Station 131 at separation to Station 122 at engine burn-out, a distance of nine inches. This center-of-gravity travel occurs in approximately 30 seconds.

The weight and center of gravity at engine burn-out reflect 20 lb of fuel remain in the thrust chamber.

6.4 Pressurization System Weight Comparisons

A pressurization system data summary is presented in Table 6-18 for the pressure-fed propulsion system being proposed for the Pioneer Vehicle and compared with data for a pump-fed but otherwise similar system. It will be noted that the decrease in weight of the pressurization system for the pump-fed propulsion system is about 184 pounds. About 168 pounds of this arises from the reduction of wall thickness of the helium tank made possible by the reduction of the maximum pressure from 3000 to 335 psi. The remaining 16 pounds is an actual decrease in helium weight.

Table 6-18 also shows that an increase of weight of about 37 pounds in the pump-fed system results if an APU fuel tank of 19.8 cubic feet volume to be pressurized to 100 psi is added to the system. On the other hand, no change of weight of the pressure-fed system is necessary due to the large helium residual available between the 300 psi pressurization level and the 100 psi APU tank pressure.

Similarly, Table 6-19 shows a 72 pound decrease in pressurization system weights for LOX-JP-4 APU system if a turbopump is used in the propulsion system instead of pressure feed, and the pressurizing helium is stored at the temperature of liquid oxygen. More striking, however, is the weight comparison between helium storage at LOX temperature (-297°F) and at room (100°F assumed) temperature. The cost of not taking advantage of the opportunity for storing the pressurizing gas at LOX temperature is 125 pounds for the pressure-fed system, and 25 pounds for the pump-fed system. In both cases, the weight of the warm gas storage system is well over twice that of the cold gas storage system.

TABLE 6-1

SUMMARY: WEIGHT AND BALANCE

BREAKDOWN	Pioneer Orbiting Test Vehicle	Pioneer Visual Recon. Vehicle	Pioneer Ferret Recon. Vehicle
Structure	435	435	435
Propellant Tanks - Assembly	495	495	495
Propulsion System	163	163	163
Pressurization System	30	30	30
Guidance & Attitude Control	150	150	150
Distruct System	41	41	41
WEIGHT EMPTY	1,314	1,314	1,314
C. G. @ Sta.	(132)	(132)	(132)
PAYLOAD			
Equipment + Antennae	494	537	590
Power Source	537	494	441
Propellant	1,155	1,155	1,155
TOTAL PAYLOAD	2,186	2,186	2,186
C. G. at Sta.	(135)	(137)	(140)
GROSS WEIGHT	3,500	3,500	3,500
C. G. at Sta.	(134)	(135)	(137)
Adapter Section	-156	-156	-156
WEIGHT AT SEPARATION	3,344	3,344	3,344
C. G. at Sta.	(131)	(134)	(136)
WEIGHT AT BURNOUT	2,209	2,209	2,209
C. G. at Sta.	(122)	(126)	(129)

2386

TABLE 6-2

MOMENTS OF INERTIA

PIONEER FERRET RECONNAISSANCE VEHICLE

CONDITION	WEIGHT (LBS)	C.G. (STA)	MOMENTS OF INERTIA* PITCH OR YAW	ROLL
GROSS WEIGHT	3,500	137	1020	150
AT SEPARATION	3,344	136	930	120
AT BURNOUT	2,209	129	820	110

---

* SLUG FT^2.

TABLE 6-3

PAYLOAD: PIONEER ORBITAL TEST VEHICLE

ITEMS	WEIGHT
EQUIPMENT	
1. Beacon Transponder	20
2. Command Receiver	8
3. Telemetering	40
4. Instrumentation	400
Antennae	
1. Slot (Beacon)	1
2. Slot (Command)	1
3. Slot-Circumferential (Tele.)	12
4. Slot- " (Space)	12
Total Equipment Weight and C. G.	494 lb at Sta. 170
POWER SOURCE	
1. Batteries (Sta. 57)	307
2. Batteries (Sta. 90)	200
3. Circuitry	30
Total Power Source Weight and C. G.	537 lb at Sta. 72
PROPELLANT	
1. Fuel - UDMH	304
2. Oxidizer - WFNA	851
Total Propellant Weight and C. G.	1,155 lb at Sta. 151
TOTAL PAYLOAD WEIGHT AND C. G.	2,186 lb at Sta. 135

TABLE 6-4

PAYLOAD: PIONEER VISUAL RECONNAISSANCE VEHICLE

ITEMS	WEIGHT
EQUIPMENT	
1. Container - Sealed (Incl.)	150
Camera	
Film Processor	
Photo Readout Pickups	
2. TV Readout Electronics	150
3. TV Modulator + Data Transmitter	30
4. Command Receiver	8
5. Programmer	30
6. Power Converter	50
7. Beacon Transponder	20
8. Telemetering - Monitor	45
Antennae	
1. Slot - Circum (Monitor)	12.0
2. Slot - (Command)	1.0
3. Slot - (Beacon)	1.0
4. Dish 36" Assembly (Incl.)	40.0
Dish - 36" - 10#	
Transmitter 10#	
Gimbal & Mts. 10#	
Servo Motor 10#	
Total Equipment Weight and C. G.	537 lb at Sta. 170
POWER SOURCE	
1. Batteries (Sta. 57)	274
2. Batteries (Sta. 90)	200
3. Circuitry	20
Total Power Source Weight and C. G.	494 lb at Sta. 72
PROPELLANT	
1. Fuel - UDMH	304
2. Oxidizer - WFNA	851
Total Propellant Weight and C. G.	1,155 lb at Sta. 151
TOTAL PAYLOAD WEIGHT AND C. G.	2,186 lb at Sta. 137

TABLE 6-5

PAYLOAD: PIONEER FERRET RECONNAISSANCE VEHICLE

ITEMS	WEIGHT
EQUIPMENT	
1. Crystal Video Receivers (Ferret)	150
2. Data Reduction Equip.	30
3. Data Recorder + Electronics	150
4. Data Transmitters (Incl. Spares)	8
5. Command Receiver (Incl. Spares)	8
6. Programmer	10
7. Power Converter	50
8. Beacon Transponder	20
9. Telemetering - (Monitor)	45
Antennae	
1. Dish - 35" (Incl. Mts. & Fittings)	13
2. Dish - 25" " " " "	8
3. Dish - 17" " " " "	4
4. Dish - 11" " " " "	2
5. Helix - 11" Dia.Coil (Incl.Mts. & Fittings)	30
6. Helix - 5" Dia.Coil " " "	18
7. Helix -2.5" Dia.Coil " " "	18
8. Slot - Circum. (Tele-Monitor)	12
9. Slot - Circum. (Data-Ferret)	12
10. Slot (Command)	1
11. Slot (Beacon)	1
Total Equipment Weight and C. G.	590 lb at Sta. 176
POWER SOURCE	
1. Batteries (Sta. 57)	321
2. Batteries (Sta. 90)	100
3. Circuitry	20
Total Power Source Weight and C. G.	441 lb at Sta. 66
PROPELLANT	
1. Fuel - UDMH	304
2. Oxidizer - WFNA	851
Total Propellant Weight and C. G.	1,155 lb at Sta. 149
TOTAL PAYLOAD WEIGHT AND C. G.	2,186 lb at Sta. 140

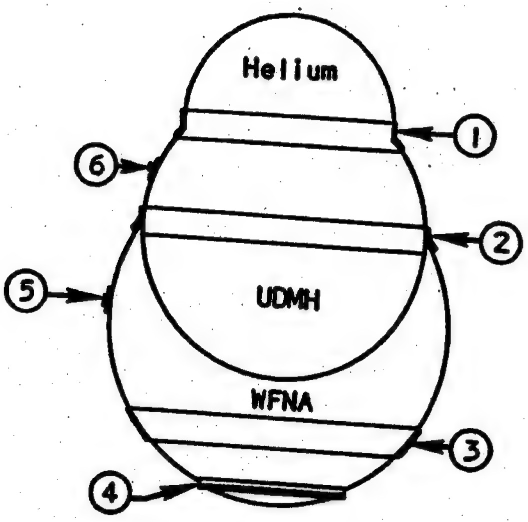
TABLE 6-6

WEIGHT EMPTY STRUCTURE

ITEMS	WEIGHT
Nose Section	
1. Hoist Assy.	4
2. Skin (.020 Stl)	4
3. Skin (.080 Mag)	56
4. Insulation	21
5. Ring	5
6. Frames & Doublers	2
7. Access Door Instl.	1
8. Misc.	2
Total Nose Sec. Weight & C. G.	91 lb at Sta. 77
Skirt Section - Fixed	
1. Skin (.080 Mag)	25
2. Separation Clamp - (Upper Sec.)	42
3. Tension Strap	32
4. I - Sections	2
5. Ring	16
6. Misc.	1
Total Fixed Weight & C. G.	118 lb at Sta. 142
Adapter Section	
1. Skin (.080 Mag)	67
2. Ring	6
3. Ring	6
4. Ring	14
5. Separation Clamp (Lower Sec.)	53
6. I - Sections	5
7. Doublers	2
8. Track Instl.	2
9. Misc.	1
Total Adapter Sec. Weight & C. G.	156 lb at Sta. 173
Supports	
1. Equipment - Fwd.	39
2. Equipment - Aft.	27
3. Guide Instl.-Adapter	5
Total Support Weight and C. G.	71 lb at Sta. 120
Total Structure Weight and C. G.	435 lb at Sta. 136

TABLE 6-7

PROPELLANT TANKS ASSEMBLY



ITEMS	WEIGHT
1. Sphere - He.	198
2. Sphere - UDMH	57
3. Sphere - WFNA	93
4. Ring ①	11
5. Ring ②	33
6. Tension Ring ③	10
7. Cover - Access ⑤	4
8. Cover - Access ⑥	4
9. Insulation - He. - 1.0"	3
10. Insulation - UDMH - 1.0"	7
11. Insulation - WFNA - 1.0"	12
12. Ring & Cover ④	60
13. Misc.	3
Total Assembly Weight & C. G.	495 lbs. @ Sta. 128

TABLE 6-8

PROPULSION SYSTEM

	ITEMS		WEIGHT LBS.
1.	Thrust Chamber		90 *
2.	Mount - Thrust Chamber		8 *
3.	Valves & Regulators		21 *
4.	Electrical Sequence Units		7 *
5.	Plumbing		7 *
6.	Control Engines (Incl. Valves)	(2)	16 *
7.	Feed Line - UDMH		3
8.	Feed Line - WFNA		1
9.	Filler Valve & Plumbing - UDMH		2
10.	Filler Valve & Plumbing - WFNA		3
11.	Supts. - Control Engines		3
12.	Misc.		2
	Total System Weight & C. G. (* Aerojet Items.)		163 lbs. @ Sta. 202

TABLE 6-9

PRESSURIZATION SYSTEM

	ITEMS		WEIGHT LBS.
1.	Pressure Regulators	(3)	1.3
2.	Check Valves	(3)	0.7
3.	Relief Valves	(3)	1.5
4.	Solenoid Valves	(3)	3.3
5.	Pressure Operated Valve	(1)	0.4
6.	Helium (9.18 ft.^{3} @ 3000 psi)		17.4
7.	Bladder Instl.	(2)	1.6
8.	Plumbing & Fittings		3.1
9.	Misc.		0.7
	Total System Weight & C. G.		30.0 lbs. @ Sta. 113

(See Propellant Tanks Section for He. Spheré)

TABLE 6-10

GUIDANCE AND ATTITUDE CONTROL

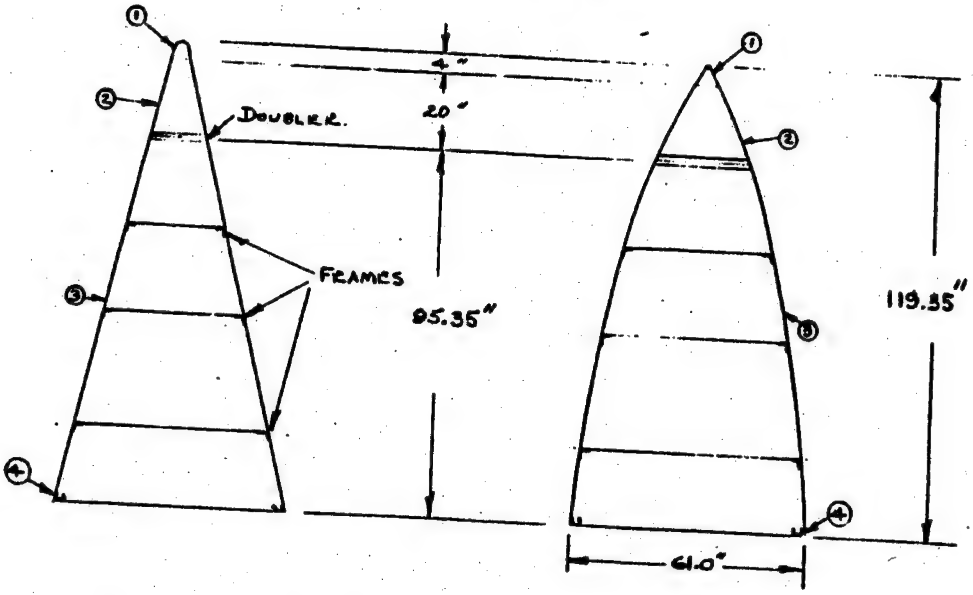
ITEMS	WEIGHT
1. Torque Drive Unit	20
2. Autopilot	20
3. Thrust On-Off System	5
4. Transition Computer	20
5. Attitude Reference	20
6. Damping Computer	15
7. Attitude Indication	10
8. Horizon Sensory System	5
9. Torque Wheel Trim System	15
10. Container	15
11. Circuitry	5
Total System Weight & C. G.	150.0 lbs. @ Sta. 73

TABLE 6-11

DESTRUCT SYSTEM

ITEMS	WEIGHT
Charge	28.5
Mechanical Mechanism	12.5
Total System Weight & C. G.	41.0 lbs. @ Sta. 115

Apdx. A Table 6-12
Weight Comparison for Nose Section
(Conical vs. Ogive)



Iems	Conical Nose		Ogive Nose	
	Mtl. & Gage	Weight	Mtl. & Gage	Weight
Hoist Fitting ①		4		4
Skin ②	.020 Stl.	4	.020 Stl.	5
Skin ③	.070 Mag.	52	.080 Mag.	76
Ring ④		5		5
Ring ⑤				
Frames & Doublers		2		3
Insulation.		21		28
Comparative Weight		88		121.

TABLE 6-14

PROPELLANT TANK STUDY

(He - UDMH - WFNA)

ITEMS	HIGH PRESSURE (PRESSURE FED)		LOW PRESSURE (PUMP FED)	
	MTL* GAGE	WEIGHT	MTL* GAGE	WEIGHT
Sphere - He (32.0" OD)	.215	198	.032	30
Sphere - UDMH (44.2" OD)	.036	57	.016	24
Sphere WFNA (54.0" OD)	.045	93	.016	33
Ring (Junction He & UDMH Spheres)		11		3
Ring (Junction UDMH & WFNA Spheres)		33		4
Pump - Engine		0		45
Helium		16.4		0.5
Comparative Totals		408.4		139.5

* Mtl. AM 350 Stainless

Ref:

	Pressures - PSI	
	High	Low
He Sphere	3,000	335
UDMH "	315	35
WFNA "	300	30

Apdx. A Table 6-15
Summary: Propellant Tank Study
(JP-4, LOX)

FIGURE - I

TANK		PRESSURE* HIGH	PRESSURE* LOW
APU	-	330	100
JP-4	-	315	35
LOX	-	300	30

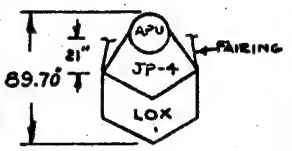


FIGURE - II

TANK		PRESSURE* HIGH	PRESSURE* LOW
APU	-	330	100
JP-4	-	315	35
LOX	-	300	30

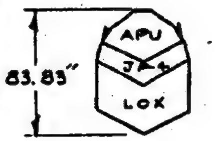


FIGURE - III

TANK		PRESSURE* HIGH	PRESSURE* LOW
APU	-	330	100
JP-4	-	315	35
LOX	-	300	30

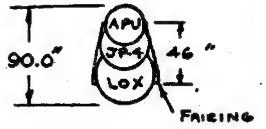
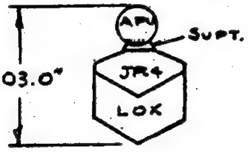


FIGURE - IV

TANK		PRESSURE* HIGH	PRESSURE* LOW
APU	-	100	100
JP-4	-	315	35
LOX	-	300	30



TANKS	FIGURE - I				FIGURE - II				FIGURE - III				FIGURE - IV			
	HI.-PRESS. WEIGHT		LOW PRESS. WEIGHT		HI.-PRESS. WEIGHT		LOW PRESS. WEIGHT		HI.-PRESS. WEIGHT		LOW-PRESS. WEIGHT		HI PRESS. WEIGHT		LOW-PRESS. WEIGHT	
	.012 MIN	.016 MIN	.012 MIN	.016 MIN	.012 MIN	.016 MIN	.012 MIN	.016 MIN	.012 MIN	.016 MIN	.012 MIN	.016 MIN	.012 MIN	.016 MIN	.012 MIN	.016 MIN
APU	47	47	17	23	255	255	198	198	58	58	17	23	17	23	17	23
JP-4	252	252	45	51	193	193	37	40	72	72	29	36	230	230	59	59
LOX	249	249	45	51	251	251	45	51	130	130	42	52	249	249	44	51
TANK WT.	548	548	107	125	699	699	280	289	260	260	88	111	496	496	120	133
FAIRING-SUPT	17	17	17	17	0	0	0	0	39	39	39	39	10	10	13	13
RINGS	324	324	132	132	484	484	204	204	53	53	7	7	321	521	192	192
	889	889	256	274	1183	1183	484	493	352	352	134	157	1027	1033	325	338

MTL. AM-350
PRESSURE - PSIG

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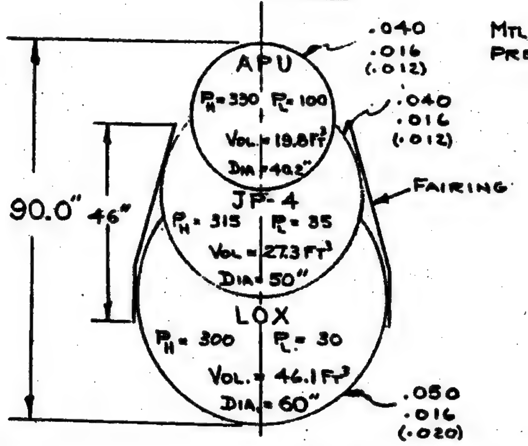
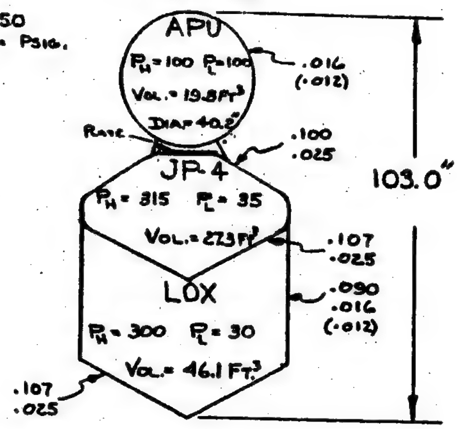
Apdx. A Table 6-16
Propellant Tank Study
(JP-4, LOX)

FIG. - III



MTL. AM 350
PRESSURE - PSIG.

FIG. - IV



TANKS	FIGURE - III						FIGURE - IV					
	HIGH PRESSURE			LOW PRESSURE			HIGH PRESSURE			LOW PRESSURE		
	MTL. GAGE	WEIGHT .012 MIN.	WEIGHT .016 MIN.	MTL. GAGE	WEIGHT .012 MIN.	WEIGHT .016 MIN.	MTL. GAGE	WEIGHT .012 MIN.	WEIGHT .016 MIN.	MTL. GAGE	WEIGHT .012 MIN.	WEIGHT .016 MIN.
APU												
SPHERE	.040	58	58	.012 .016	17	23	.012 .016	17	23	.012 .016	17	23
JP-4												
SPHERE	.040	72	72	.012 .016	29	36	-	-	-		-	-
PLATE	-	-	-	-	-	-	.600	44	44	.190	14	14
FRUSTUM	-	-	-	-	-	-	.100	71	71	.025	18	18
CONE	-	-	-	-	-	-	.107	115	115	.025	27	27
LOX												
SPHERE	.050	130	130	.012 .016	42	52	-	-	-	-	-	-
CYLINDER	-	-	-	-	-	-	.090	134	134	.012 .016	17	24
CONE	-	-	-	-	-	-	.107	115	115	.025	27	27
TOTAL TANKS		260	260		88	111		496	502		120	133
FAIRING-SUPT.		39	39		39	39		10	10		13	13
RINGS		53	53		7	7		521	521		192	192
TOTAL WT.		352	352		134	157		1027	1033		325	338

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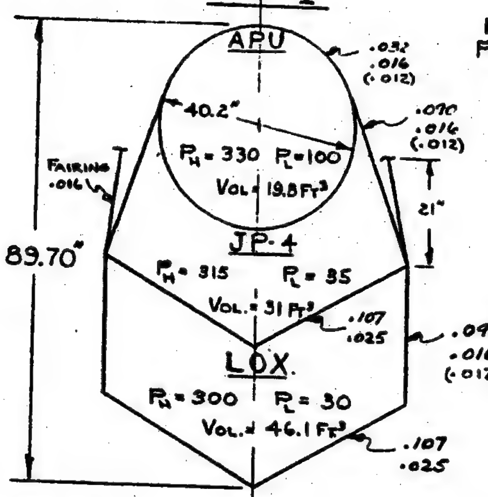
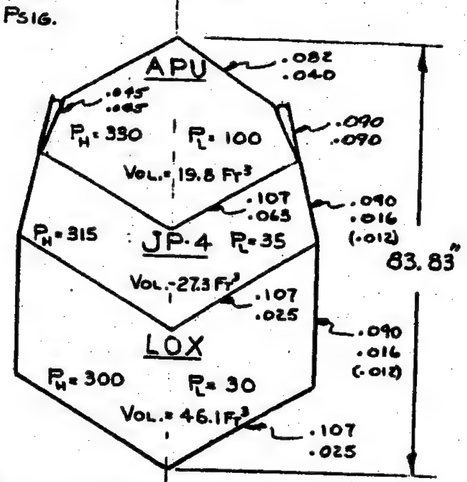
Apdx. A Table 6-17
Propellant Tank Study
(JP-4, LOX)

FIG - I



MTL. = AM350
PRESSURE - PSIG.

FIG. - II



TANKS	FIGURE - I						FIGURE - II					
	HIGH PRESSURE			LOW PRESSURE			HIGH PRESSURE			LOW PRESSURE		
	MTL. GAGE	WEIGHT		MTL. GAGE	WEIGHT		MTL. GAGE	WEIGHT		MTL. GAGE	WEIGHT	
		.012 MIN.	.016 MIN		.012 MIN	.016 MIN		.012 MIN	.016 MIN		.012 MIN.	.016 MIN.
APU												
SPHERE	.032	47	47	.012 .016	17	23						
CONE-U	-	-	-	-	-	-	.082	40	40	.040	20	20
FRUSTUM	-	-	-	-	-	-	.045 .090	125	125	.045 .090	125	125
CONE-L	-	-	-	-	-	-	.107	90	90	.063	53	53
JP-4												
FRUSTUM	.090	137	137	.012 .016	18	24	.090	77	77	.012 .016	10	13
CONE	.107	115	115	.025	27	27	.107	116	116	.025	27	27
LOX												
CYLINDER	.090	134	134	.012 .016	18	24	.090	135	135	.012 .016	18	24
CONE	.107	115	115	.025	27	27	.107	116	116	.025	27	27
TOTAL-TANKS		548	548		107	125		699	699		280	289
FAIRING-SUPT.		17	17		17	17		0	0		0	0
RINGS		324	324		132	132		484	484		204	204
TOTAL WT.		889	889		256	274		1183	1183		484	493

Table 6-18

PRESSURIZATION SYSTEM DATA SUMMARY

System	Pressure-fed		Pump-fed	
Propellant tank (Note 1)	Fuel	Oxid	Fuel	Oxid
Tank Volume, cu ft	19.6	29.0	19.6	29.0
Tank Pressure, psi	315	300	35	30
He Sphere pressure, psi (Note 2)	3000		335	
He Sphere volume, cu ft	9.12		9.12	
He Sphere inside diam., in.	31.0		31.0	
He Sphere weight, lb	198		30	
Helium gas weight, lb (Note 3)	16.4		0.5	
Total weight, lb.	214.4		30.5	

Addition of an APU tank of 19.8 cu ft volume pressurized at 100 psi would cause the following changes:

He Sphere pressure, psi	No change (See Note 4)	730
He Sphere volume, cu ft	"	9.12
He Sphere inside diam., in.	"	31.0
He Sphere weight, lb	"	66
Helium gas wt., lb	"	1.3
Total weight, lb	"	67.3

NOTES:

(1) Fuel is unsymmetrical-dimethyl hydrazine; oxidizer is white fuming nitric acid. Total propellant weight 3560 lb.

(2) Initial helium temperature assumed 100°F.

(3) Helium assumed to be heated to 0°F. after being withdrawn from storage vessel.

(4) Residual in sphere after propellant tanks are pressurized at 300 psi is more than sufficient to pressurize the APU tank at 100 psi.

Table 6-19

PRESSURIZATION SYSTEM DATA SUMMARY

System	Pressure Fed			Pump Fed		
Propellant tank (Note 1)	Fuel	Oxid	APU	Fuel	Oxid	APU
Tank volume, cu ft	27.3	43.1	19.8	27.3	43.1	19.8
Tank pressure, psi (Note 2)	315	300	100	25	30	100
He Sphere volume, cu ft (Note 3)	2.90			0.50		
He Sphere inside diam., in.	21.2			11.8		
He Sphere weight, lb	72.5			12.5		
Helium gas weight, lb	15.4			2.7		
Total weight, lb	87.9			15.2		

NOTES: (1) Fuel is JP-4; oxidizer is LOX. Total propellant weight 4340 lb. APU fuel is 1230 lb hydrazine.

(2) Helium assumed to be heated to 100°F between storage sphere and propellant tank.

(3) Helium storage pressure 3000 psi, temperature -297°F (temperature of liquid oxygen). For He at 3000 psi and 100°F.

	Pressure Fed	Pump Fed
He Sphere Vol., cu ft	8.57	1.73
He Sphere I.D., in.	30.4	17.1
He Sphere wt., lb	178	38.0
Helium gas wt., lb	15.4	2.7
Total weight, lb	193.4	40.7

7. TEMPERATURE CONTROL OF SATELLITE COMPONENTS

This analysis indicates that the average temperature of the equipment in the pioneer satellite vehicle will be in the vicinity of room temperature (20° C) if certain emission conditions of the outer surface of the vehicle can be maintained. If no heat is generated internally, room temperature can be obtained by painting the inside surfaces black and by processing the outer vehicle surface so that the ratio of the absorption coefficient for low temperature radiation to the absorption coefficient for the solar spectrum is about 0.77. Polished nickel has approximately this ratio of absorption coefficients. Slightly different surface emissivities would have to be chosen to attain room temperature if heat is generated within the vehicle.

An auxiliary means of controlling the average temperature, however, will have to be provided because of the possible variation of the average internal heat dissipation and because of the gradual erosion of the surface. This means could consist of (1) heating or cooling a fluid reservoir by using an external radiator only during the "day" or during the "night", or (2) properly drawing or retracting highly reflective metal foils about the internal components.

For an average internal power dissipation of 50 watts, the average temperature of the components will be about 5° C if the outer surface is black, and about 28° C if the outer surface has the emission properties of polished nickel (an absorption coefficient of 0.12 for low temperature radiations and 0.15 for the solar spectrum). Because of the large heat capacity of the components within the vehicle, the maximum "day-night" difference of their average temperature will be only about 5° C for a darkened outer surface, and about 2° C for the nickel type surface. With special effort, the variation of the temperature of the film processing unit can be held within about ± 0.5° C.

A great deal of care must be taken to design the equipment so that the required temperature range for each component can be maintained. This analysis assumes that a good thermal design has already been accomplished and that a fluid is circulated about the equipment to maintain temperature control. Fluid circulation jackets are assumed to surround the equipment units completely so that the surfaces will be radiating heat to each other and to the skin of the vehicle at nearly the same temperature.

The average temperature of the equipment will depend upon (1) the heat generated by the equipment, (2) the emissivities of the radiating surfaces, and (3) the heat absorbed by the skin of the vehicle from the external incident radiation.

Heat radiation from the sun and from the earth incident on the surface elements of the vehicle has been calculated as a function of the orbital angle by assuming a nose-up attitude of the vehicle and using an albedo of 0.43 and a solar constant of 1340 watts $/m^2$. The earth was assumed to radiate from its entire surface the energy that it absorbs from the sun. The altitude of the vehicle was taken as 300 miles. If the external surface of the vehicle is "coated white" such that it has an absorption coefficient for the solar spectrum of 0.30 and a low temperature absorption coefficient of 0.80, then the temperatures of the surface elements in radiative equilibrium with the absorbed external radiation are given as a function of the orbit angle θ in Figs. 7-1, 7-2 and 7-3.

Of course, the actual temperature differences of the skin elements around the vehicle will be much more relaxed because of (1) radiation from the skin to the interior of the vehicle, (2) heat conduction along the skin, and (3) heat generation within the vehicle. A maximum difference in the vicinity of about 30° C should be expected for the conditions assumed in this analysis. The temperature of the skin will be taken such that

$$E_{20} \, 6 \, T_2^4 \, A_2 = H(\theta) + H_{12}, \qquad (1)$$

where

E_{20} = low temperature emissivity of outer skin surface

A_2 = area of skin

6 = Stefan-Boltzmann constant

$H(\theta)$ = total external heat radiation per unit time absorbed by skin while at the orbital angle θ, and

H_{12} = net heat per unit time transferred from heat generating components to skin.

Let A_1 be the area of the heat generation components that is available for direct radiation to the skin. Then, if H_1 and H_2 are respectively the luminosities of surfaces A_1 and A_2, the heat per unit time leaving A_1 is A_1H_1, and the heat per unit time radiating

[redacted]

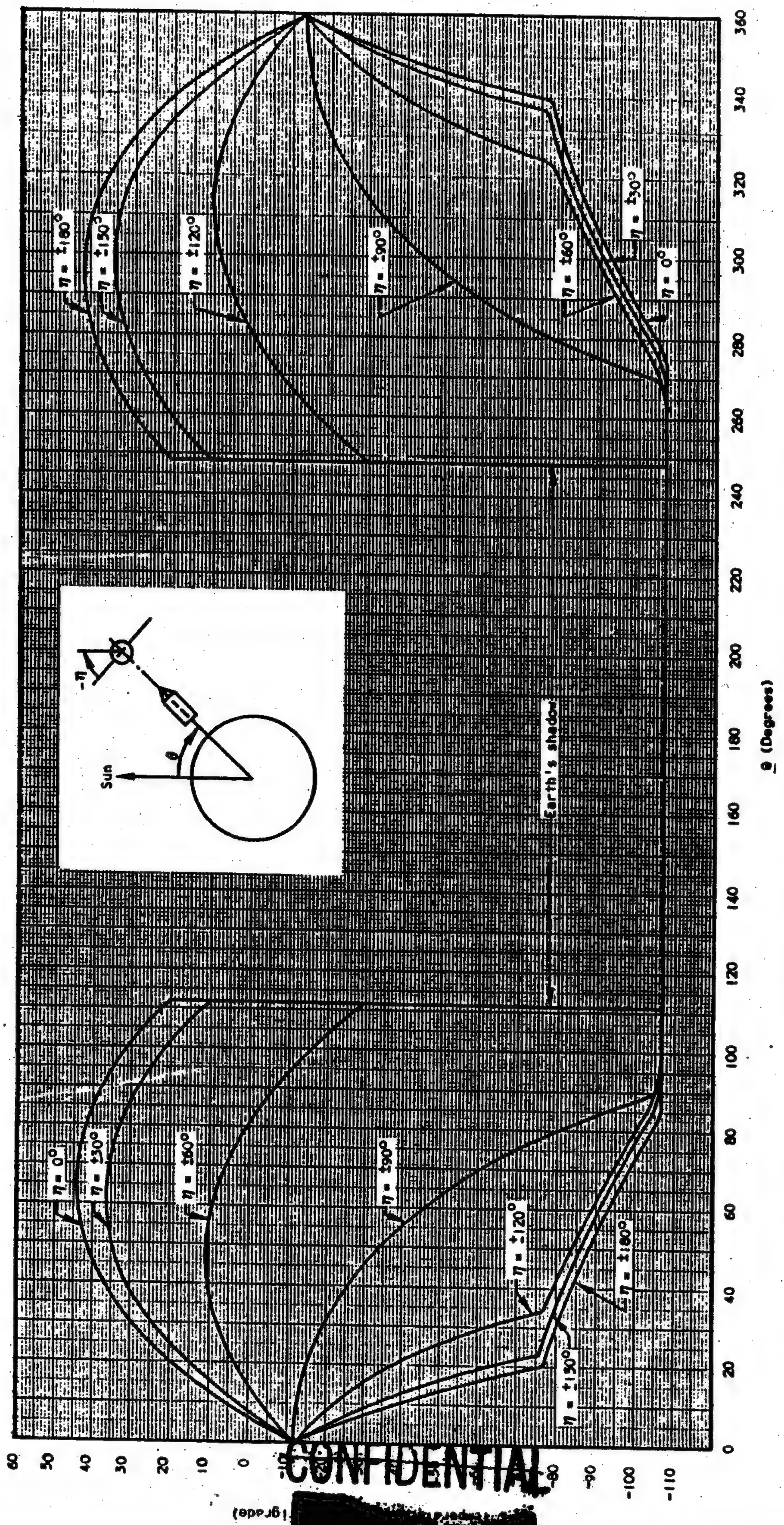


Fig. 7-1 Adiabatic Skin Temperature of Conical Surface vs. Position in Orbit

[redacted]

[redacted]

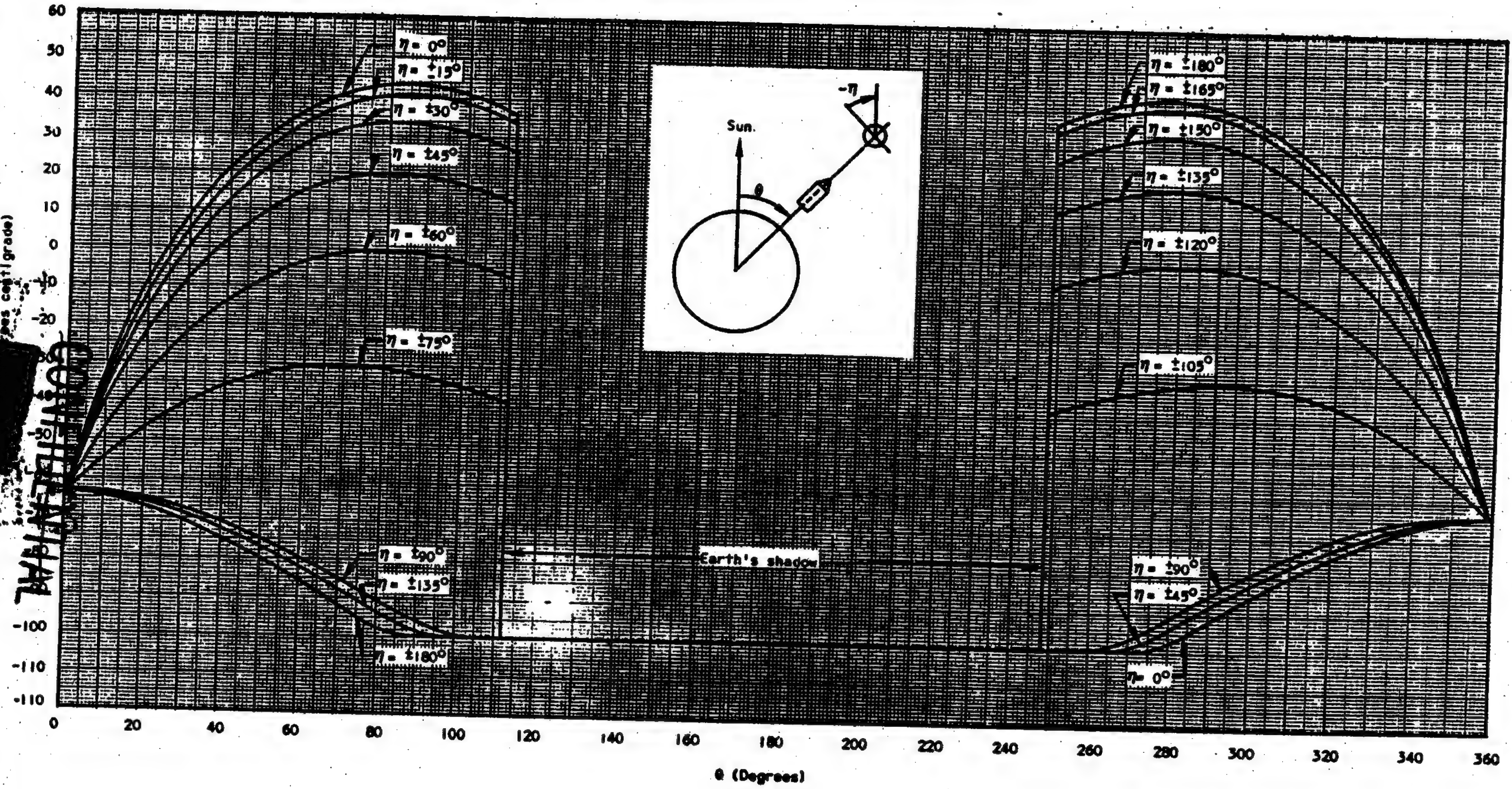


Fig. 7-2 Adiabatic Skin Temperature of Cylindrical Surface vs. Position in Orbit

[redacted]

[redacted]

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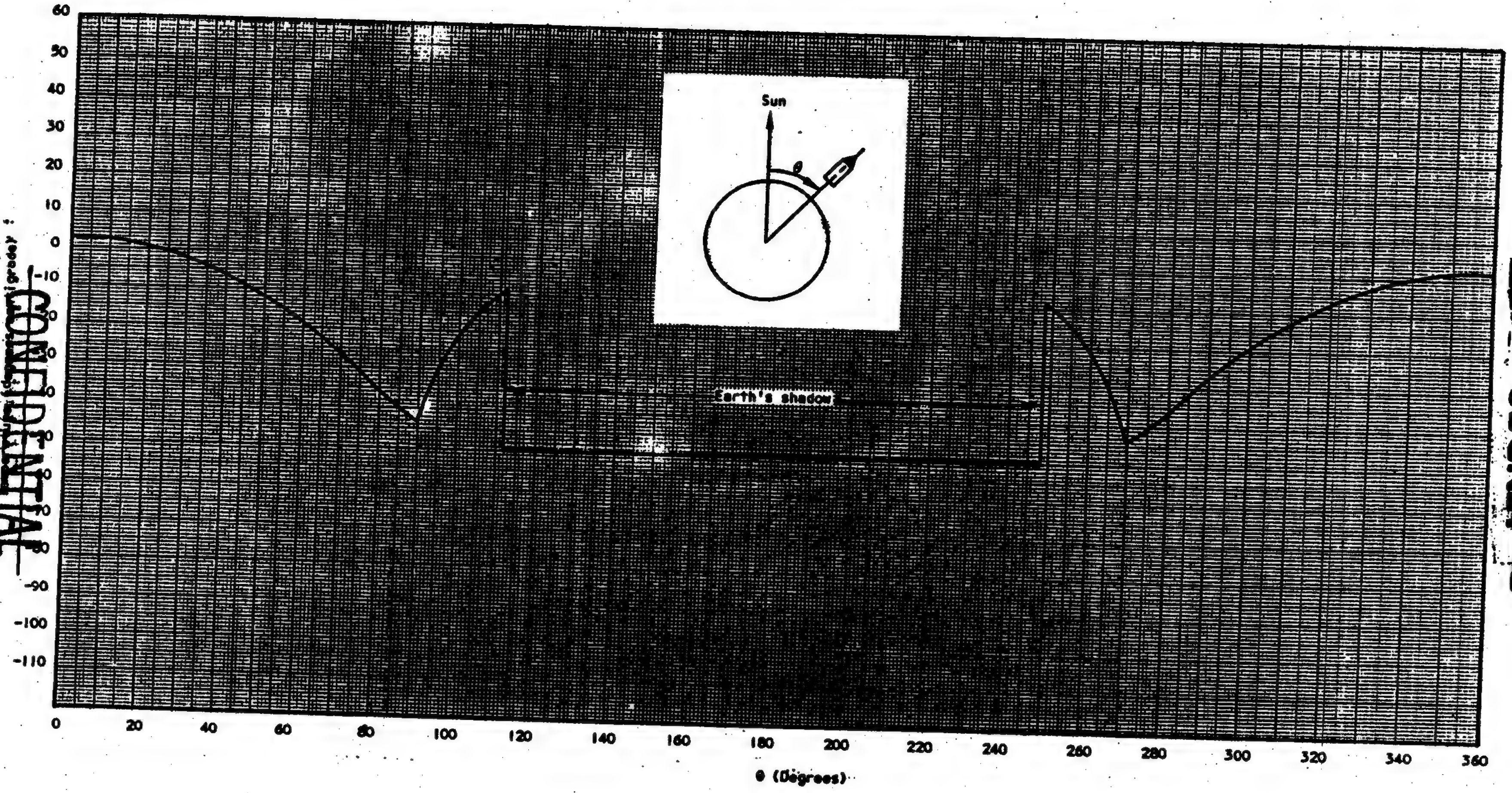

Fig. 7-3 Adiabatic Skin Temperature of Bottom Surface vs. Position in Orbit

[redacted]

internally from surface A_2 is A_2H_2. Of the quantity A_2H_2, however, only $\frac{A_1}{A_2}$ (A_2H_2) * reaches A_1, the remainder reaches other portions of A_2 itself. Hence, the net heat flow from A_1 is

$$H_{12} = A_1(H_1 - H_2).$$

If the average temperature of A_1 is taken to be T_1, then H_1 equals the heat radiation at the temperature T_1 plus the reflected portion of H_2. H_2 equals the heat radiation at temperature T_2 plus the reflected portion of the intensity $\frac{A_1}{A_2}$ H_1 incident on A_2, plus the reflected portion of the intensity $(1- \frac{A_1}{A_2})H_2$ incident on A_2 from other regions of A_2 itself. Or, if E_1 and E_2 represent, respectively, the emissivity of A_1 and the inner surface of A_2, then

$$H_1 = E_1\ 6T_1^4 + (1-E_1)\ H_2, \text{ and}$$

$$H_2 = E_2\ 6T_2^4 + (1-E_2)\ \frac{A_1}{A_2}\ H_1 + (1-E_2)\ (1- \frac{A_1}{A_2})H_2.$$

By solving the above two equations for H_1 and H_2, the net heat transfer from A_1 becomes

$$H_{12} = A_1(H_1 - H_2) \quad \frac{A_1\ 6(T_1^4 - T_2^4)}{\frac{1}{E_1} + \frac{A_1}{A_2}\left[\frac{1}{E_2} - 1\right]} \qquad (2)$$

and, of course

$$H_{12} = -Ms\ \frac{dT_1}{dt} + P \qquad (3)$$

---

* This is an approximation.

where

M = Mass of heat generating components

s = Average specific heat of M

P = Heat generated in M per unit time, and

T_1 has been taken as the average temperature of M.

Finally, by substituting the expression for T_2^4 from equation (1) into equation (2), and then equating the resultant expression to the right side of equation (3), the following equation is obtained giving the temperature change of the mass M:

$$Ms \frac{dT_1}{dt} = \frac{H(\theta)}{K A_2 E_{20}} - \frac{6 T_1^4}{K} + P, \qquad (4)$$

$$\text{where } K = \frac{1}{A_2 E_{20}} + \frac{1/A_1}{1/E_1 + \frac{A_1}{A_2}\left[\frac{1}{E_2} - 1\right]}$$

Since the fore and aft sections of the vehicle will be fairly well isolated thermally by the rocket fuel tanks (this condition can be insured by polishing the tank walls), heat balance equations should be set up separately for each section. Only the nose section is considered in this analysis. For this section, the following values will be used:

$M = 500$ lb

$S = 0.4$ Cal $/{}^{\circ}C/g$ (5)

$A_1 = 5.00\ M^2$

$A_2 = 8.35\ M^2$

$E_1 = E_2 = E_{20} = 0.95$

$K = 0.311$

In the steady state condition, after the influence of the initial component temperatures becomes small, the average temperature of the equipment can be obtained by substituting $\frac{dT_1}{dt} = 0$ in equation (4) and and using the average values of $H(\theta)$ and P during an orbital period. Thus,

$$6\,\overline{T}_1^{\,4} = \frac{\overline{H(\theta)}}{A_2 E_{20}} + K P ,$$

If we assume that only attitude control equipment and batteries are located in the nose, then the average heat generated will be about 50 watts. By using $\overline{H(\theta)} = 2620$ watts, the average temperature of the components will be 278° K (5° C). If the surface of the vehicle is prepared so that it has a low temperature absorption coefficient of 0.12 and a coefficient of 0.15 for the solar spectrum, then $H(\theta) =$ 407 watts, and the average temperature of the components would be 28° C. Since the average power dissipation in the aft section of the vehicle will not be much greater than heat generation in the nose, the average temperature of the equipment in that section should be nearly the same.

Equation (4) gives a maximum "day-night" temperature difference of about 5° C for a darkened vehicle outer surface and about 2° C for the polished surface discussed above.

The temperature of the film processing unit must be kept at a certain known temperature ±0.5° C while the film is being developed.

This condition can be met by placing the pressurized compartment, which contains the camera, the film, the processing unit, and the read-out table, within an enclosure of area A_2. This enclosure will then be maintained at an average temperature of about 20° C by the circulating fluid described above. Although the temperature of this fluid will vary by a maximum of ±2.5° C within the orbital period, the average temperature of the pressurized unit will only vary by about ±0.5° C as shown below because of its heat capacity.

Equations (2) and (3) also apply to this configuration, so that the average temperature of the pressurized unit will have the following time variation:

$$Ms \frac{dT_1}{dt} = P - A_1 E_{12} 6(T_1^4 - T_2^4), \qquad (6)$$

where

$$E_{12} = \frac{1}{\frac{1}{E_1} + \frac{A_1}{A_2} \left[\frac{1}{E_2} - 1 \right.}$$

and

Ms = heat capacity of pressurized unit

T_1 = average temperature of mass M

T_2 = temperature of enclosure

P = heat generated in pressurized unit

A_1 = area of pressurized unit

A_2 = area of enclosure

The average temperature T_o of the pressurized unit is given by the expression,

$$\sigma T_o^4 = \sigma \bar{T}_2^4 + \frac{P}{A_1 E_{12}} \tag{7}$$

where $\bar{P}$ = average power generated per orbital period,

and $\bar{T}_2$ = average temperature of enclosure.

By putting $T_1 = T_1 + \tau$ and $T_2 = \bar{T}_2 + \Delta T_2$, then the solution of equation (6) is approximately:

$$\tau \approx \left[P - \bar{P} + 4A_1E_{12}\,\sigma\,\bar{T}_2^3\,\Delta T_2\right]\frac{t}{Ms} \tag{8}$$

The following values will be used for this configuration:

$$Ms = 3.52 \times 10^4\ \frac{\text{watt-seconds}}{^{o}C}, \quad \left[\begin{array}{l}\text{6 liters of solutions}\\ \text{6 lb of film}\\ \text{15 lb aluminum}\end{array}\right.$$

$A_1 = 0.617\ M^2$

$A_2 = 0.926\ M^2$

$$P = \left[\begin{array}{l}\text{30 watts for 10 minutes while taking pictures}\\ \text{40 watts for 5 minutes while developing film}\\ \text{40 watts for 6 minutes for movable table during read-out}\end{array}\right.$$

$\bar{P} = 8.22$ watts

$\bar{T}_2 = 293^\circ K\ (20^\circ C)$

$E_1 = E_2 = 0.95$

$E_{12} = 1.087$

By substituting the appropriate values into equation (7), the average temperature of the pressurized unit equals 21° C.

From the calculations indicated above, ΔT_2 should undulate from $+2.5$ C to -2.5° C in a period of 90 minutes. Thus, the maximum variation in the temperature of the pressurized unit from the mean can be estimated from equation (8) by putting $P = \overline{P}$, $\Delta T_2 = \frac{5^\circ}{\pi}$ C, and $t = 3340$ sec. (duration of "day"). The result is $\tau = 0.58^\circ$ C. If the film processing unit is kept in neighborhood of the center of the pressurized unit, the temperatures fluctuations should be somewhat less severe .

This analysis of the gross heat effects indicates that the emissivity of the surface of the vehicle can be chosen so that the average temperature of the vehicle components will be in the vicinity of normal room temperature. Moreover, the environment of the film processing unit should be adequate if the average temperature of the circulating fluid can be maintained at a constant value.

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